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NO. OF 225

NAVAIR 01-10FAB-1

PRELIMINARY

NATOPS FLIGHT MANUAL

NAVY MODEL

F-111B

AIRCRAFT

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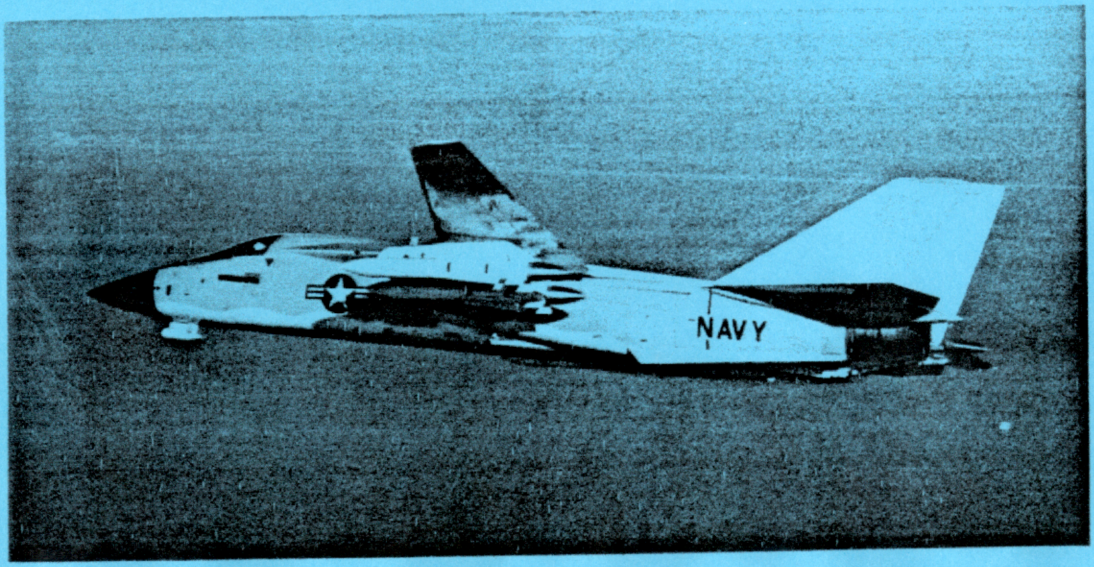

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AIRCRAFT 1
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SAFETY SUPPLEMENT SUMMARY

The following list contains: the previously cancelled or incorporated Supplements the outstanding Safety Supplements, if any; and the Safety Supplements incorporated in this issue. In addition, space is provided to list those Safety Supplements received since the last issue.

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NUMBER	SUBJECT OR DISPOSITION

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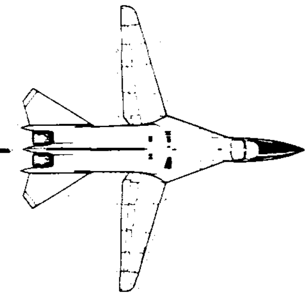


* SEE SUPPLEMENTAL NATOPS FLIGHT MANUAL NAVAIR 01-10FAB-1A FOR ADDITIONAL DATA

26512-1/1-0

YOUR FIRST F-111B FLIGHT MANUAL

26512-1/99-0

SCOPE

This Flight Manual contains descriptive and procedural information based on the latest available information necessary for safe and efficient operation of the Navy Model F-111B aircraft, Bu. No. 152714 and subsequent. It should be noted that this is a Preliminary Manual and shall be replaced by a regular NATOPS Manual at a later date.

SOUND JUDGEMENT

This manual provides the best available operating instructions for most circumstances, but no manual is a substitute for sound judgement. Multiple emergencies, adverse weather, or terrain may require modification of the procedures contained herein. You can handle emergencies best if you know everything about your aircraft.

ARRANGEMENT

This manual is divided into eleven sections.

SECTION I - AIRCRAFT

Part 1 - GENERAL DESCRIPTION

Dimensions, cockpit layout, instrument panels.

Part 2 - SYSTEMS

Descriptive breakdown of system and system controls combined, including normal and emergency procedures.

Part 3 - AIRCRAFT SERVICING

Minimum turning radius, starting requirements, danger areas and power requirements.

Part 4 - AIRCRAFT OPERATING LIMITATIONS

Limitations and restrictions that shall be observed for safe and efficient operation of the engines and the airframe.

SECTION II - INDOCTRINATION

To be supplied at a later date.

SECTION III - NORMAL PROCEDURES

Part 1 - BRIEFING/DEBRIEFING

Part 2 - MISSION PLANNING

Part 3 - SHORE-BASED PROCEDURES - PILOT

Procedures from scheduling to postflight, to include night flying, FCLP and MLP.

Part 4 - CARRIER-BASED PROCEDURES - PILOT

To be supplied at a later date.

Part 5 - SHORE-BASED PROCEDURES - MCO

Part 6 - CARRIER-BASED PROCEDURES - MCO

To be supplied at a later date.

SECTION IV - FLIGHT PROCEDURES

To be supplied at a later date.

SECTION V - EMERGENCY PROCEDURES

SECTION VI - ALL-WEATHER OPERATION

Simulated and actual instruments, turbulence and thunderstorms, cold weather, tropic operations, and desert operations.

SECTION VII - COMMUNICATIONS EQUIPMENT AND PROCEDURES

Radio navigation, visual, and ground deck procedures.

SECTION VIII - WEAPONS SYSTEM

A description and discussion of the integrated attack navigation system equipment.

SECTION IX - FLIGHT CREW COORDINATION

To be supplied at a later date.

SECTION X - NATOPS EVALUATION

To be supplied at a later date.

SECTION XI - PERFORMANCE DATA

NATOPS POCKET CHECK LIST

The NATOPS Pocket Check List (NAVAIR 01-10FAB-1B) provides, in abbreviated form, essential information for operation of the F-111B. This Check List may be obtained in the same manner as the NATOPS Flight Manual. Changes to it are concurrent with, and dated the same as the NATOPS Flight Manual.

UPDATING THE MANUAL

This manual will maintain current with additional information and updated procedures, through an active program of changes and revisions, prepared and distributed periodically.

SAFETY SUPPLEMENTS

Information pertaining to safety-of-flight of the F-111B will be issued by safety supplements to the NATOPS Flight Manual, until such time as Flight Manual Interim Changes (FMIC's) are promulgated by CNO and NASC.

The safety supplement summary in your Flight Manual should be checked to determine the status of existing supplements. After completion of the instructions, safety supplements shall be retained in front of the flyleaf of the manual, unless the safety supplement contains authorization to discard the page.

OPERATIONAL SUPPLEMENTS

Information pertaining to changes to operating procedures for the F-111B will be issued by operational supplements to the NATOPS Flight Manual, until such time as Flight Manual Interim Changes (FMIC's) are promulgated by CNO and NASC.

Procedures for handling operational supplements are the same as for safety supplements.

SAFETY AND OPERATIONAL SUMMARIES

The safety and operational supplement summaries are provided for the purpose of maintaining a complete record of all safety and operational supplements issued to the manual. Each time the manual is revised, the supplemental summaries will be updated to indicate disposition or incorporation or both of previously issued supplements. When a regular change is received, supplemental summaries should be checked to ascertain that all outstanding safety and operational supplements have been either incorporated or cancelled; those not incorporated should be re-noted as applicable.

WARNINGS, CAUTIONS, AND NOTES

The following definitions apply to "Warnings," "Cautions," and "Notes" found throughout the manual.

WARNING

Operating procedures, practices, conditions, etc., which may result in injury or death, if not carefully observed or followed.

CAUTION

Operating procedures, practices, conditions, etc., which if not strictly observed, may damage equipment.

Note

An operating procedure, conditions, etc., which it is essential to emphasize.

CHANGE SYMBOLS

Revised text is indicated by a black vertical line in either margin of the page, like the one printed next to this paragraph. The change symbol shows where there has been a change. The change might be material added or information restated.

WORDING

The concept of word usage and intended meaning which has been adhered to in preparing this Manual is as follows:

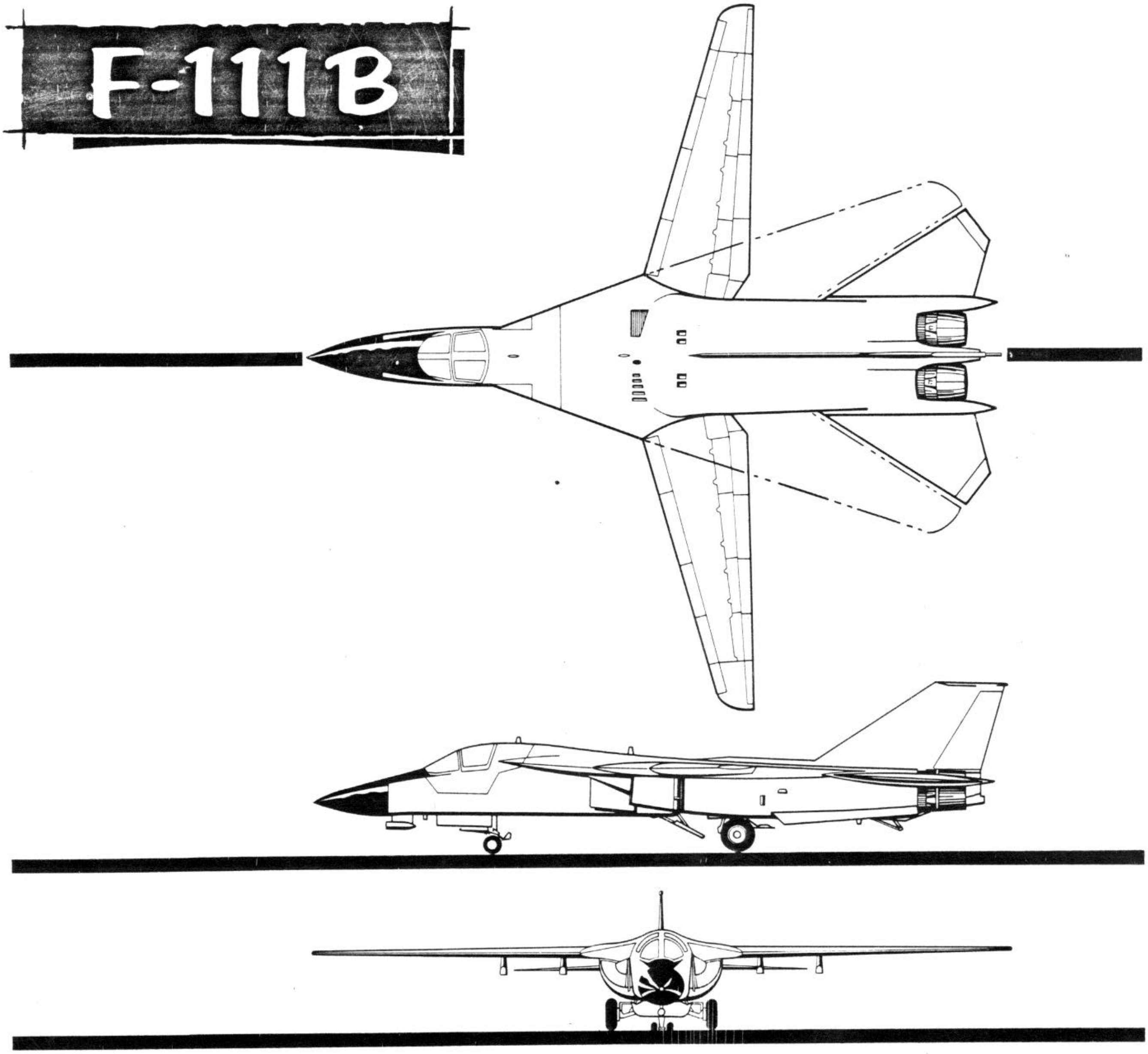
"Shall" has been used only when application of a procedure is mandatory.

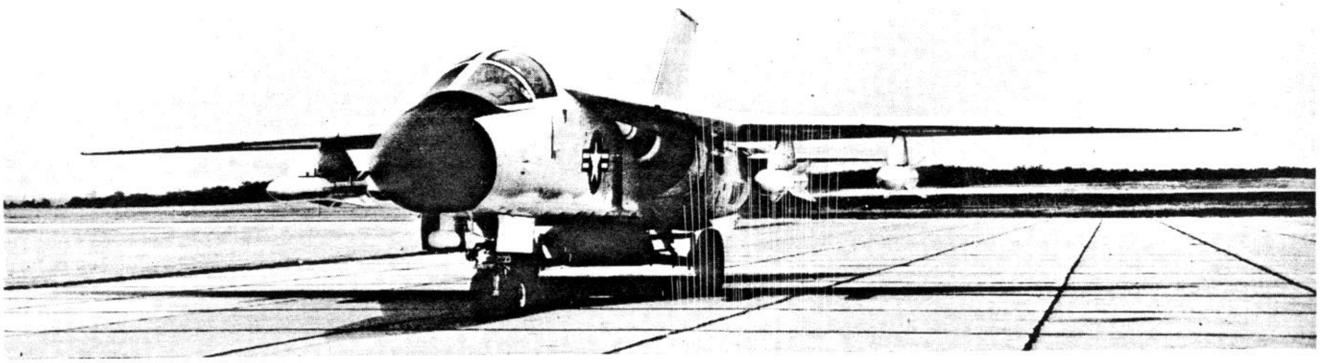
"Should" has been used only when application of a procedure is recommended.

"May" and "need not" have been used only when application of a procedure is optional.

"Will" has been used only to indicate futurity, never to indicate any degree of requirement for application of a procedure.

F-111B





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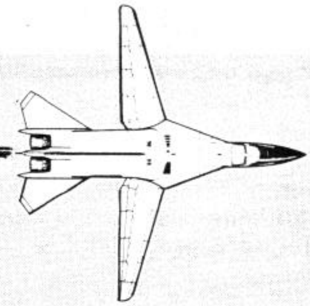
section I
AIRCRAFT



part 1
GENERAL DESCRIPTION

part 2
SYSTEMS

part 3
AIRCRAFT SERVICING

part 4
AIRCRAFT OPERATING LIMITATIONS

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part 1

GENERAL DESCRIPTION

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AIRCRAFT

The F-111B is a two place (side-by-side) high performance, all weather, long endurance, supersonic fighter aircraft. As a weapon system the primary mission is fleet defense and distant air superiority through the airborne missile (Phoenix) control system. A secondary mission capability provides ground support attack with either air-to-surface missiles, conventional armament or special weapons. Mission versatility and tactical flexibility are enhanced through independent operational capability or integration under existing tactical data systems. Thrust is provided by two TF30-P-12, axial flow, dual-compressor turbo-fan engines equipped with afterburners. The wings, equipped with leading edge slats and trailing edge flaps, may be varied in sweep, area, camber, and aspect ratio, by the selection of any wing sweep angle between 16 and 72.5 degrees. A selective forward wing sweep with high-lift devices extended provides low speed takeoff and landing capabilities. For all other regimes the wing sweep angle can be varied to optimize performance and thereby enhance airplane versatility within the designed operating envelope. The empennage consists of a fixed vertical tail with rudder for directional control, and horizontal tails which move symmetrically for pitch control and differentially for roll control. Stability and command augmentation features are incorporated in the triple electronic redundant flight control system which enhance system reliability. The tricycle-type forward retracting landing gear is hydraulically operated. The main landing gear consists of a single common trunnion upon which two wheels are singly mounted, and contains but one extending-retracting-locking system which ensures symmetrical main gear operation. Also ground loads imposed upon the gear are directional to the locked position. Stores are carried in a fuselage-enclosed weapons bay and externally on pivoting and fixed wing-mounted pylons. The fuel system incorporates both inflight and single point ground refueling capabilities.

AIRCRAFT DIMENSIONS

The overall dimensions of the aircraft are as follows:

Length, overall	68 feet, 9.5 inches
Length, radome folded, overall	64 feet, 9.2 inches
Height, overall, at vertical fin tip	16 feet, 7.7 inches
Radome folding height	14 feet, 9.3 inches
Radome folded height	13 feet, 3.0 inches
Wing span, spread, 16°	70 feet, 0.0 inches
Wing span, swept 72.5°	33 feet, 11.0 inches
Horizontal tail span	29 feet, 4.0 inches
Vertical tail height	16 feet, 7.7 inches

Refer to Section II for turning radius and ground clearance dimensions.

GENERAL ARRANGEMENT

Figure 1-2 represents the general placement of components within the aircraft.

AIRCRAFT WEIGHT

The zero fuel/stores weight of the CLEAN aircraft is approximately 46,000 pounds. Consult the applicable Handbook of Weight and Balance for the exact weight of any particular aircraft.

CREW MODULE COCKPIT LAYOUT

The aircraft accommodates a two-man crew, consisting of the pilot and missile control officer in a side-by-side seating arrangement. Figure 1-3 through 1-8 provide typical cockpit layout.

ENGINES

The aircraft is powered by two Pratt and Whitney TF30-P-12, sixteen stage, axial flow turbofan engines equipped with afterburners (figure 1-1). The engines are mounted side-by-side in the fuselage and are interchangeable. The sea level, standard day, thrust rating of each engine is in the 12,000-pound class in military power and in the 20,000-pound class in afterburner. Provisions are made for starting the engines with an external pneumatic ground starter cart. With one engine operating, the other engine can be started by using bleed air from the operating engine. Electrical power is supplied for the engine igniter plugs by an engine-driven alternator. Each engine is supplied a flow of air through a separate inlet duct below the intersection of the wing glove and fuselage. An automatically controlled movable spike is used in each inlet duct to control airflow to the engines. Additional engine inlet air is provided during ground operation and at low airspeeds by the opening of blow-in doors on the

forward portion of each inlet. Splitter plates are used at the front of the inlet ducts to remove the low energy air from the fuselage and the lower surface of the wing glove, thus minimizing the amout of boundary layer air from disturbing engine inlet air. Air from the inlet of each engine is routed through a single duct for both the basic engine section and the fan section. Three compressor stages provide the initial pressurization of the air flowing into the engine and into the fan duct. The fan duct is a full-annular duct which directs fan air flow aft, bypassing the basic engine to join the engine airflow coming from the turbine discharge. The fan air develops a significant portion of total engine thrust. Engine air is compressed by 9 stages of the low-pressure compressor (N_1) and 7 stages of the high-pressure compressor (N_2). The air is then diffused into the combustion section which contains the combustion chambers. The turbine section of the engine consists of a single-stage turbine to drive the high-pressure compressor and a three-stage turbine to drive the low-pressure compressor. The turbines are mechanically independent of each other. High-pressure (N2) speed is indicated by a tachometer. Speed of the low-pressure compressor is not monitored except by an overspeed caution lamp. After leaving the turbine section of the engine, the air is joined with fan air in the afterburner section. Bleed air from the engine compressor is used for crew module air conditioning and pressurization; for hydraulic system and fuel tank pressurization; for hydraulic oil, electrical equipment bay, generator/constant speed drive, and ground oil cooling; and for windshield rain removal and engine vortex destroyers. Also, hot bleed air is used for spike and engine inlet guide vane anti-icing.

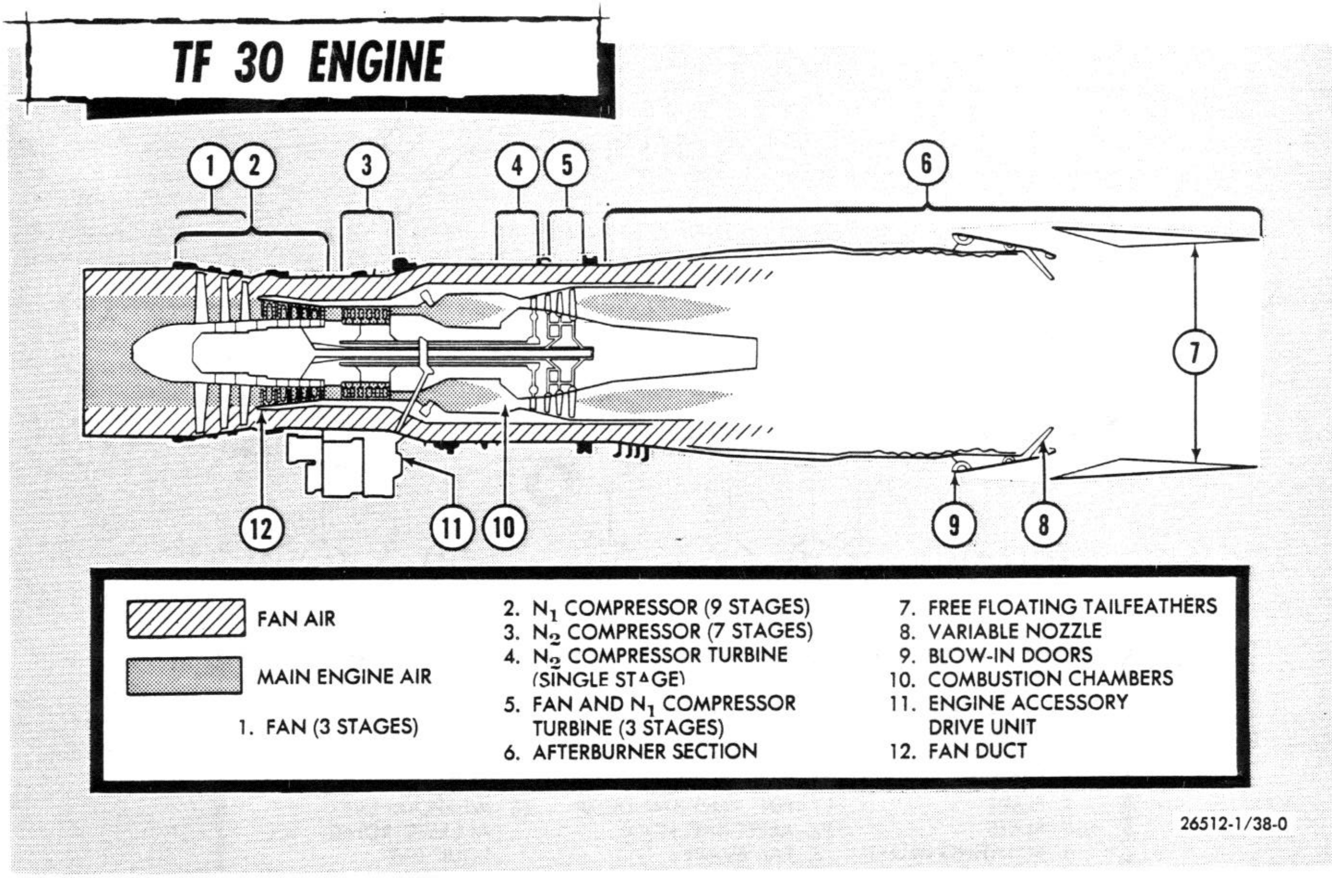
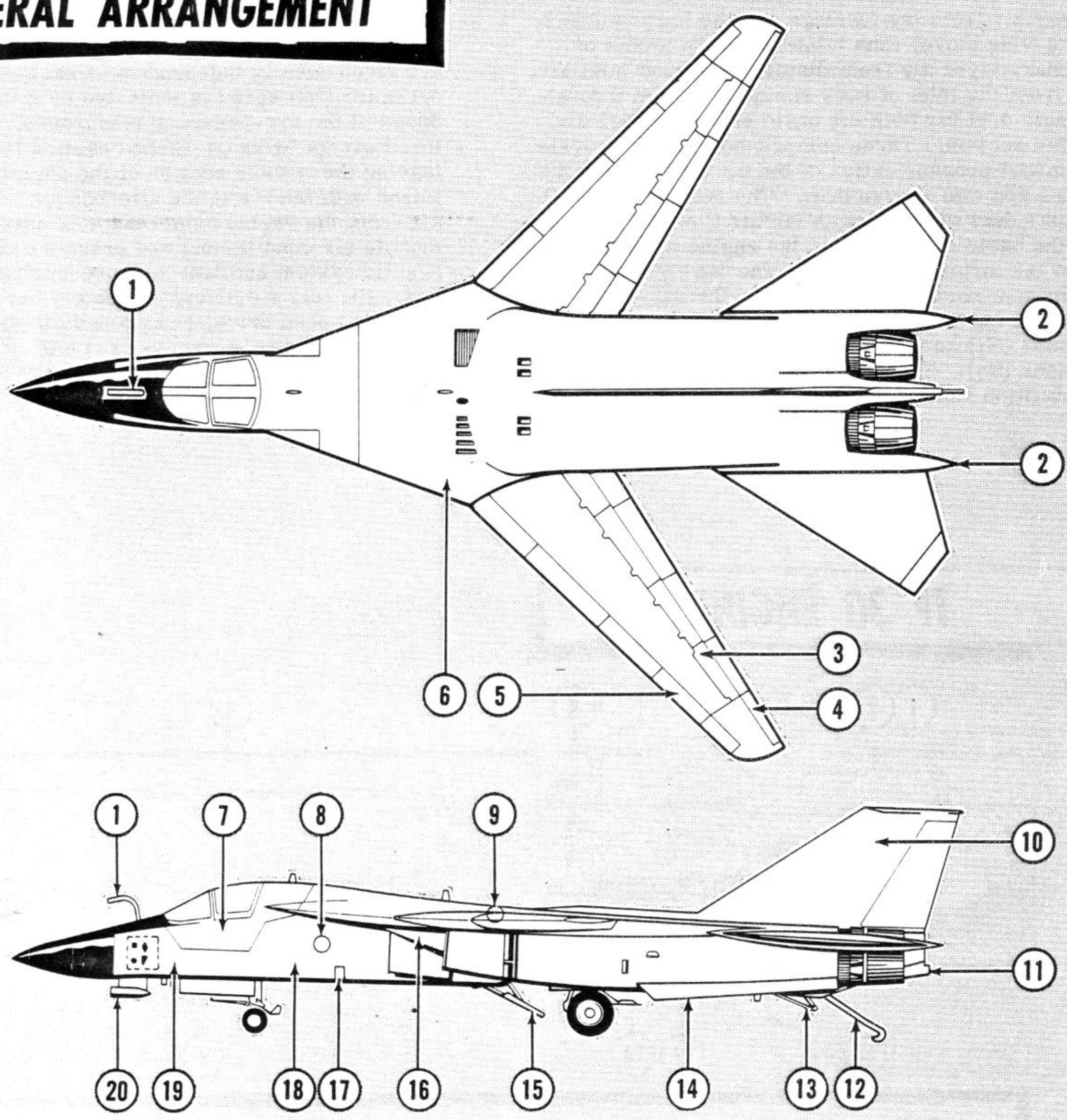


Figure 1-1

GENERAL ARRANGEMENT



1. AERIAL REFUELING PROBE EXTENDED
2. SPEED BUMPS
3. SPOILERS
4. FLAPS
5. SLATS
6. ROTATING GLOVE
7. CREW MODULE
8. SINGLE POINT - GROUND REFUEL ADAPTER
9. AIR CONDITIONING SYSTEM COOLING AIR INTAKE
10. FUEL VENT TANK
11. FUEL VENT AND DUMP
12. ARRESTING HOOK
13. TAIL BUMPER
14. STRAKE
15. SPEED BRAKE FORWARD LANDING GEAR' DR.
16. SPIKE
17. FUEL PRECHECK SELECTOR VALVE AND GAGE
18. WEAPONS BAY/ AFT ELECTRONIC EQUIP BAY
19. FORWARD ELECTRONIC EQUIPMENT BAY
20. I R DOME

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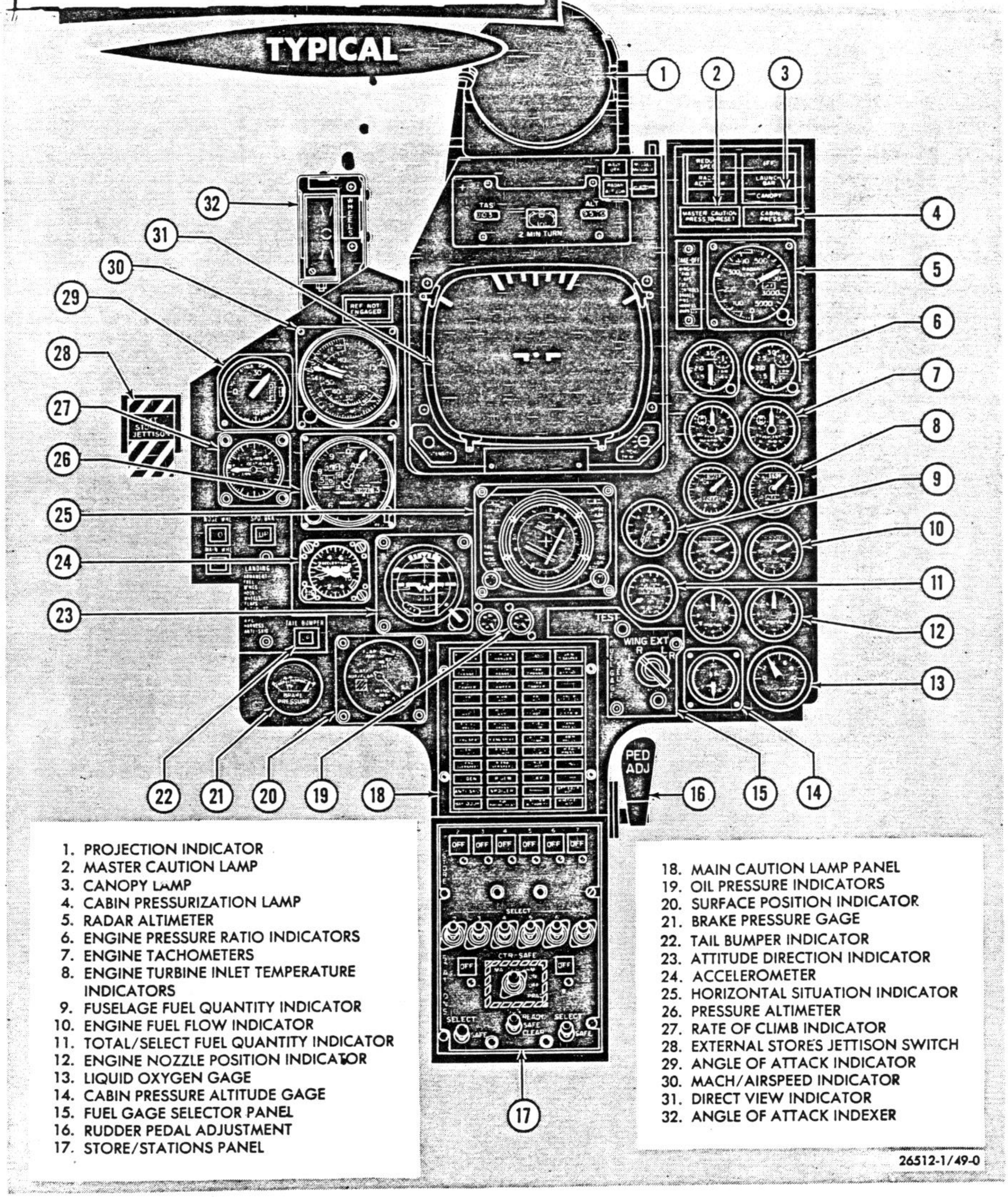
Figure 1-2

PILOT'S INSTRUMENT PANEL

TYPICAL

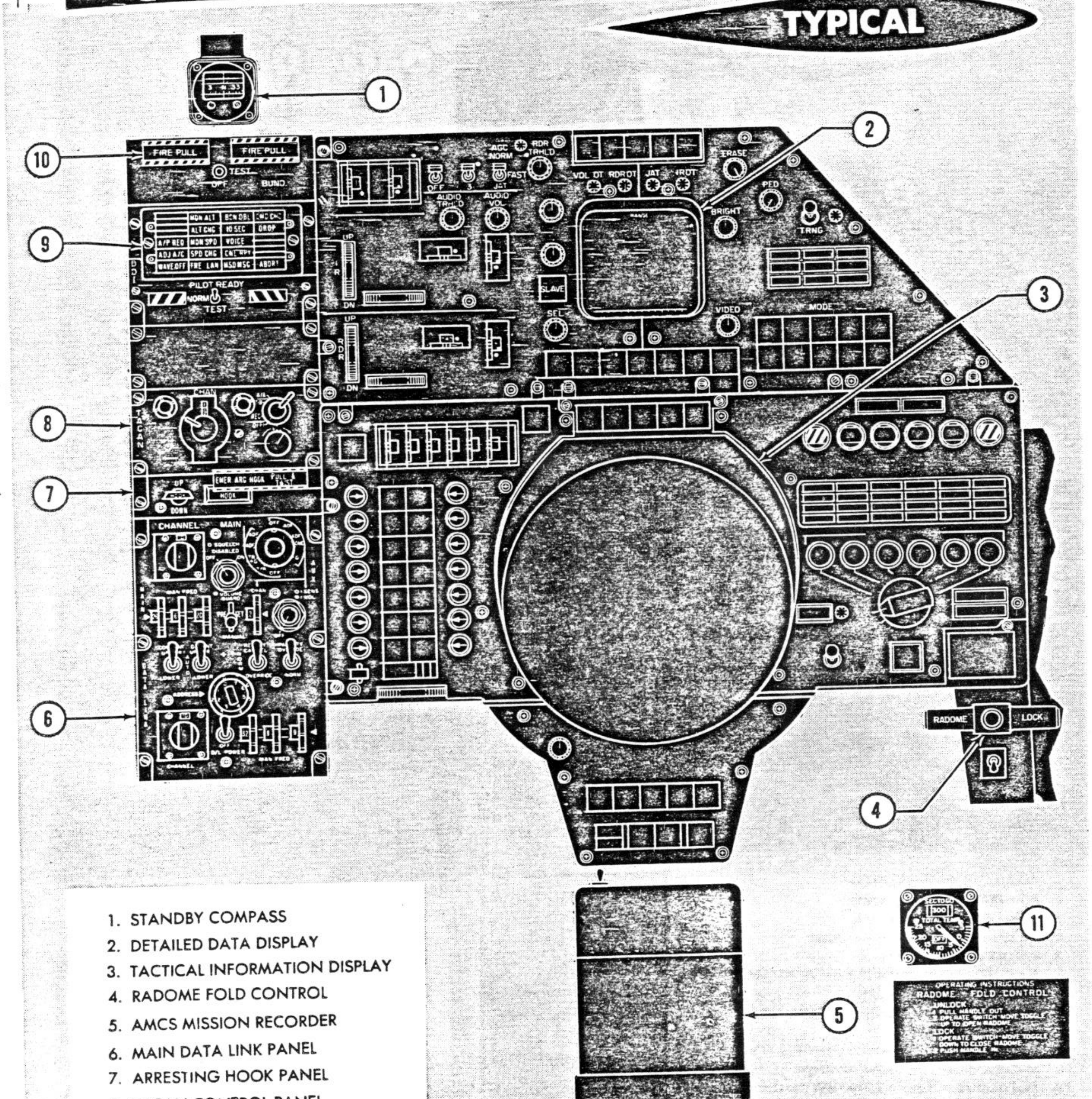


1. PROJECTION INDICATOR
2. MASTER CAUTION LAMP
3. CANOPY LAMP
4. CABIN PRESSURIZATION LAMP
5. RADAR ALTIMETER
6. ENGINE PRESSURE RATIO INDICATORS
7. ENGINE TACHOMETERS
8. ENGINE TURBINE INLET TEMPERATURE INDICATORS
9. FUSELAGE FUEL QUANTITY INDICATOR
10. ENGINE FUEL FLOW INDICATOR
11. TOTAL/SELECT FUEL QUANTITY INDICATOR
12. ENGINE NOZZLE POSITION INDICATOR
13. LIQUID OXYGEN GAGE
14. CABIN PRESSURE ALTITUDE GAGE
15. FUEL GAGE SELECTOR PANEL
16. RUDDER PEDAL ADJUSTMENT
17. STORE/STATIONS PANEL
18. MAIN CAUTION LAMP PANEL
19. OIL PRESSURE INDICATORS
20. SURFACE POSITION INDICATOR
21. BRAKE PRESSURE GAGE
22. TAIL BUMPER INDICATOR
23. ATTITUDE DIRECTION INDICATOR
24. ACCELEROMETER
25. HORIZONTAL SITUATION INDICATOR
26. PRESSURE ALTIMETER
27. RATE OF CLIMB INDICATOR
28. EXTERNAL STORES JETTISON SWITCH
29. ANGLE OF ATTACK INDICATOR
30. MACH/AIRSPEED INDICATOR
31. DIRECT VIEW INDICATOR
32. ANGLE OF ATTACK INDEXER

Figure 1-3

MISSILE CONTROL OFFICER'S INSTRUMENT PANEL

TYPICAL



1. STANDBY COMPASS
2. DETAILED DATA DISPLAY
3. TACTICAL INFORMATION DISPLAY
4. RADOME FOLD CONTROL
5. AMCS MISSION RECORDER
6. MAIN DATA LINK PANEL
7. ARRESTING HOOK PANEL
8. TACAN CONTROL PANEL
9. DIGITAL DATA INDICATOR
10. FIRE PULL HANDLES
11. TOTAL TEMP GAGE

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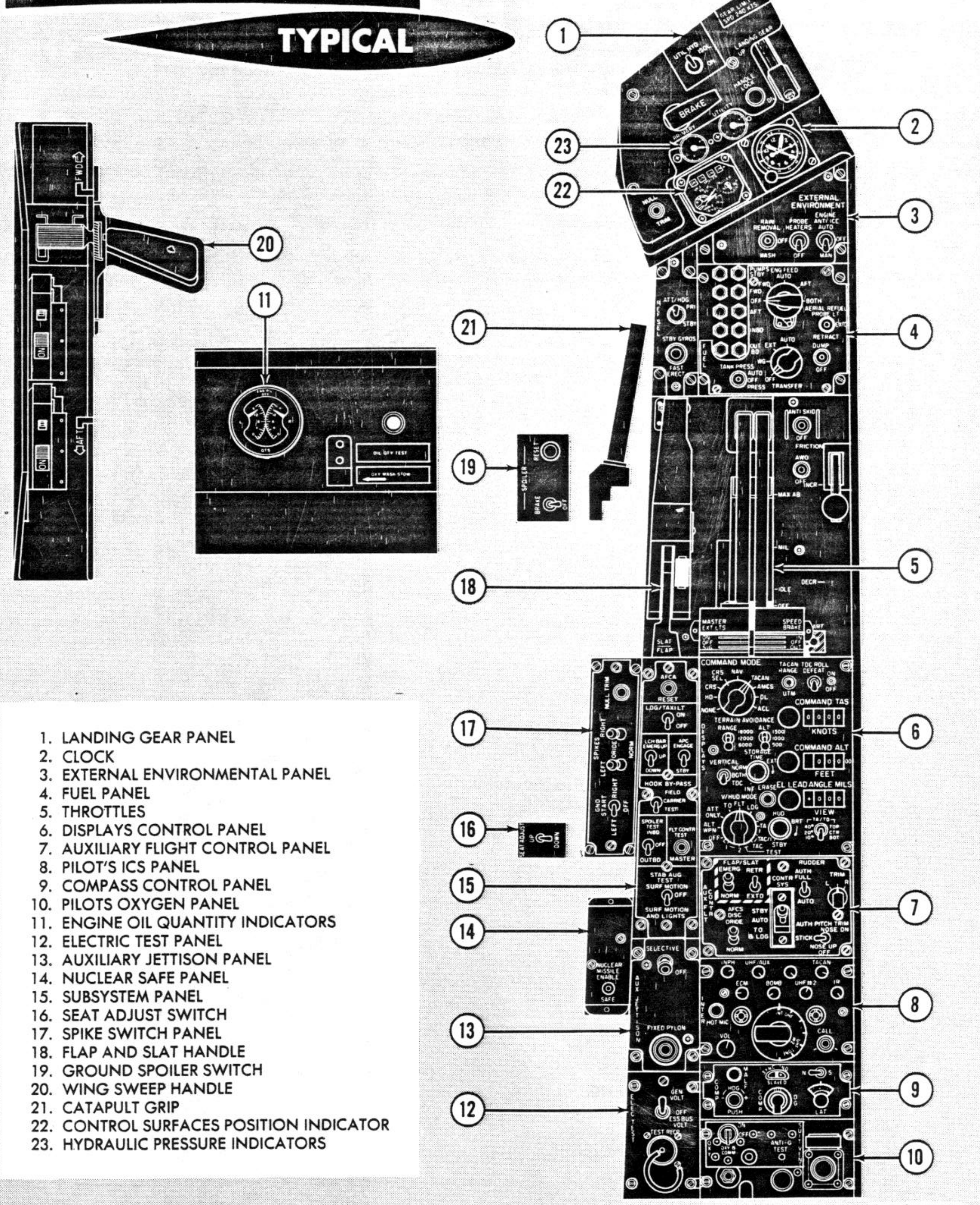
Figure 1-4

PILOT'S LEFT CONSOLE

TYPICAL



1. LANDING GEAR PANEL
2. CLOCK
3. EXTERNAL ENVIRONMENTAL PANEL
4. FUEL PANEL
5. THROTTLES
6. DISPLAYS CONTROL PANEL
7. AUXILIARY FLIGHT CONTROL PANEL
8. PILOT'S ICS PANEL
9. COMPASS CONTROL PANEL
10. PILOTS OXYGEN PANEL
11. ENGINE OIL QUANTITY INDICATORS
12. ELECTRIC TEST PANEL
13. AUXILIARY JETTISON PANEL
14. NUCLEAR SAFE PANEL
15. SUBSYSTEM PANEL
16. SEAT ADJUST SWITCH
17. SPIKE SWITCH PANEL
18. FLAP AND SLAT HANDLE
19. GROUND SPOILER SWITCH
20. WING SWEEP HANDLE
21. CATAPULT GRIP
22. CONTROL SURFACES POSITION INDICATOR
23. HYDRAULIC PRESSURE INDICATORS

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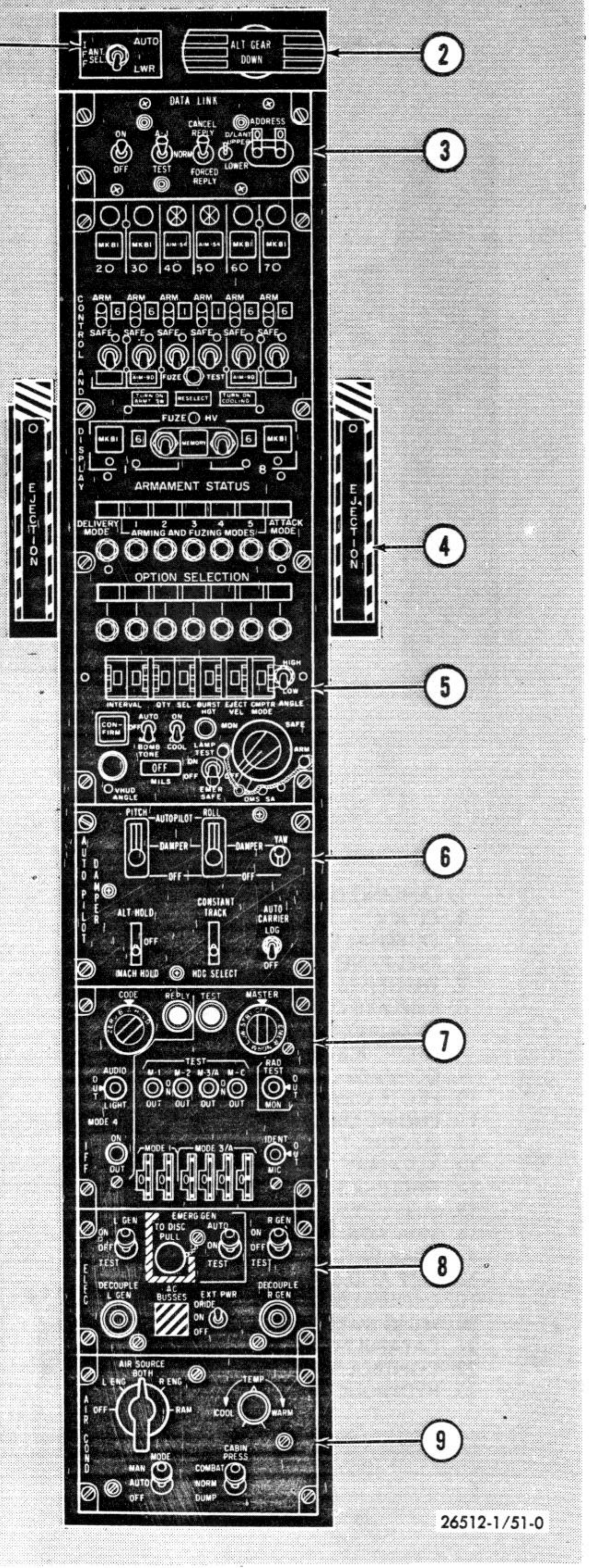
Figure 1-5

CENTER CONSOLE

TYPICAL

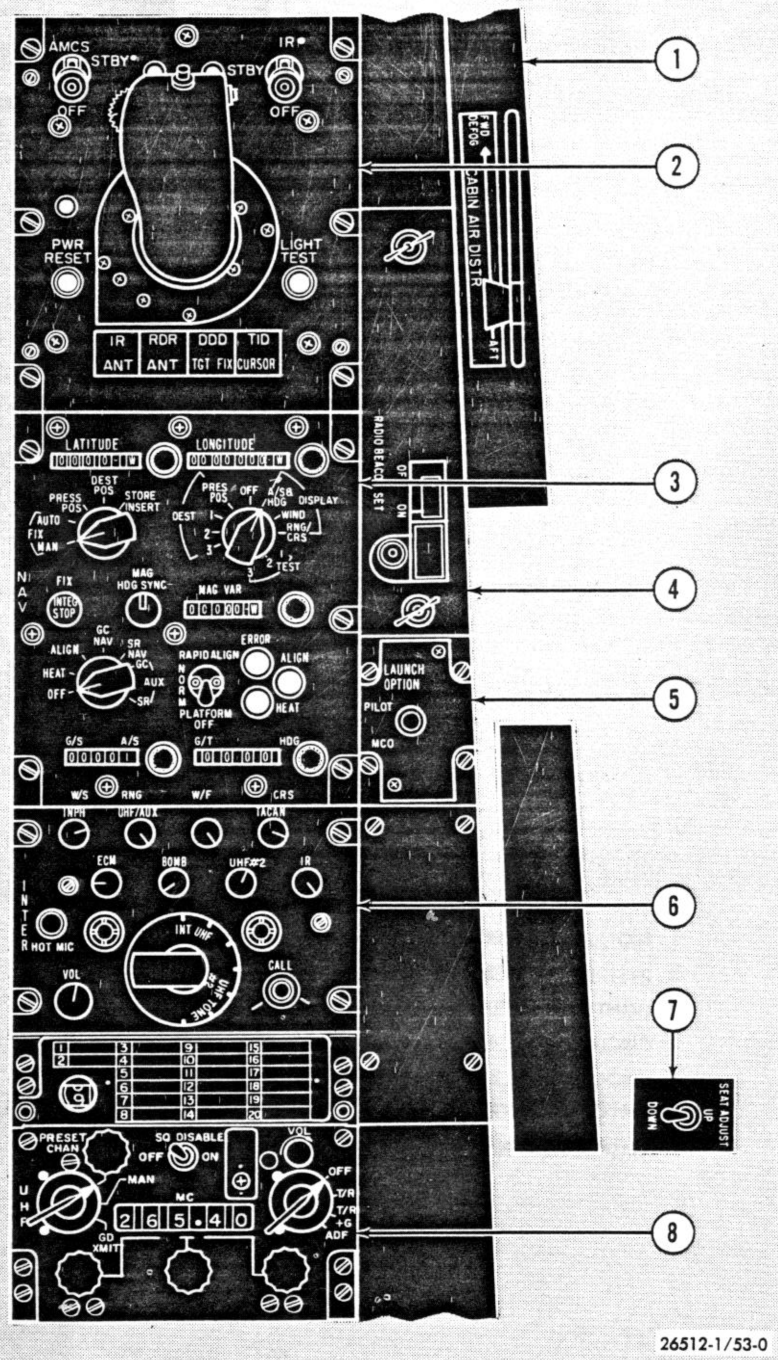


1. IFF ANTENNA SELECT SWITCH
2. ALTERNATE GEAR
3. DATA LINK REPLY PANEL
4. EJECTION HANDLES
5. CONTROL AND DISPLAY UNIT (CADU)
6. AUTOPILOT DAMPER PANEL
7. IFF CONTROL PANEL
8. ELECTRICAL
9. AIR CONDITIONING PANEL

Figure 1-6

MISSILE CONTROL OFFICER'S RIGHT CONSOLE

TYPICAL

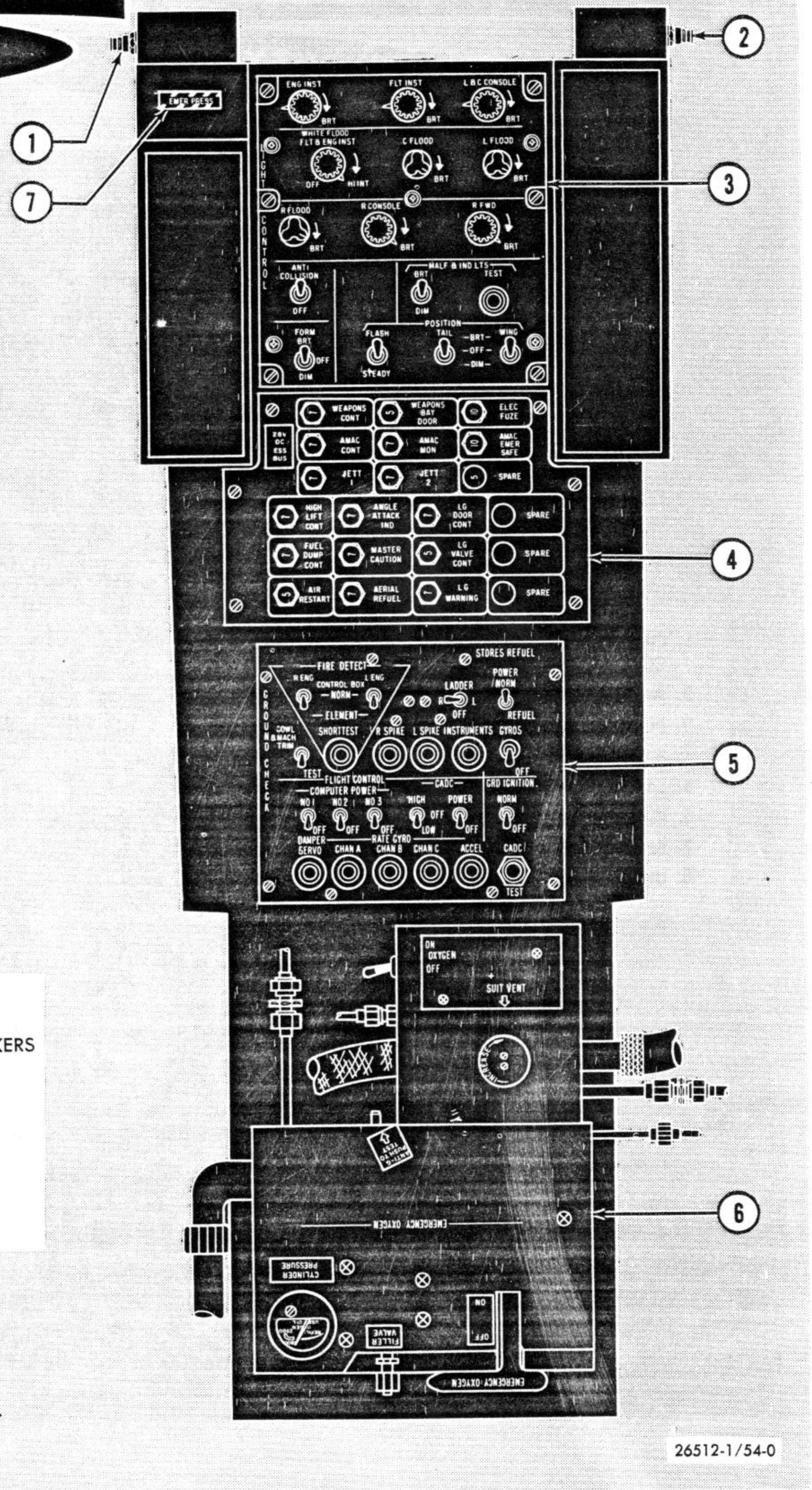


1. CABIN AIR DISTRIBUTION CONTROL PANEL
2. AMCS CURSOR CONTROL
3. NAVIGATION CONTROL PANEL
4. RADIO BECON SET
5. LAUNCH OPTION SWITCH
6. ICS CONTROL PANEL
7. SEAT ADJUST SWITCH
8. UHF RADIO PANEL

Figure 1-7

AFT BULKHEAD

TYPICAL



1. FLOOD LIGHT CIRCUIT BREAKERS
2. INSTRUMENT LIGHT CIRCUIT BREAKERS
3. LIGHTING PANEL
4. CIRCUIT BREAKER PANEL
5. GROUND CHECK PANEL
6. OXYGEN CONTROL PANEL
7. EMERGENCY PRESSURIZATION

Figure 1-8

part 2
SYSTEMS

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ENGINE FUEL SYSTEM

Each engine fuel system (figure 1-9) automatically provides optimum fuel flow for any throttle setting. This system responds to several engine operating parameters and makes it unnecessary to adjust the throttle in order to compensate for variations in inlet air temperature, altitude or airspeed. The engine fuel system consists of a two-stage, engine-driven fuel pump, fuel control unit, flowmeter, filter/heater, a pressurizing and dump valve, nozzles, and a fuel-oil heat exchanger. Fuel from the tanks is routed through the flowmeter to the centrifugal stage of the engine fuel pump, through a filter/heater, and back to the gear stage of the pump. Should these components fail, by-pass valves route fuel past the filter or first pump stage. The second pump stage delivers fully pressurized fuel to the fuel control unit which provides metered fuel flow through the fuel-oil heat exchanger to the fuel pressurizing and dump valve. This dual-function valve directs the fuel through the primary and secondary fuel manifolds to the fuel nozzles which spray the fuel into the engine combustion chambers. When the fuel pressure drops during engine shutdown, the fuel pressurizing and dump valve automatically opens and drains the primary fuel manifold.

ENGINE FUEL CONTROL UNIT

The engine fuel control unit is a hydromechanical device incorporating an engine-driven, flyball-type speed governor. The control unit consists of a fuel metering system and a computing system which schedules fuel flow to the engine as a function of throttle setting, main combustion chamber pressure, high-pressure rotor N2 speed, compressor inlet pressure, compressor inlet temperature, and flight mach number. The metering system selects the rate of fuel flow to be supplied to the engine in response to the throttle setting. However, metering sections are regulated by the fuel control computing system which monitors the various engine operating parameters. Fuel enters the fuel control through a filter which is provided with a spring-loaded bypass. Fuel metering is accomplished by maintaining a constant pressure across a variable valve area which is controlled by the computing system. The constant pressure is maintained by means of a pressure-regulating bypass valve. This valve consists of a servo-operated valve and a spring-loaded valve. Normally, the servo maintains constant valve regulation, but if the servo malfunctions, the spring valve alone will provide adequate regulation. Deviations from the desired metering pressure are sensed in the valve regulating unit, which varies the bypass flow area; the unit thereby restores the desired pressure by returning excess fuel to the pump inlet.

AFTERBURNER

The afterburner (A/B) augments engine thrust by injecting fuel into the engine exhaust stream in the afterburner section where it is ignited by a hot streak ignition system. Operation is controlled by the throttle. When the throttle is moved forward within

ENGINE FUEL SYSTEM

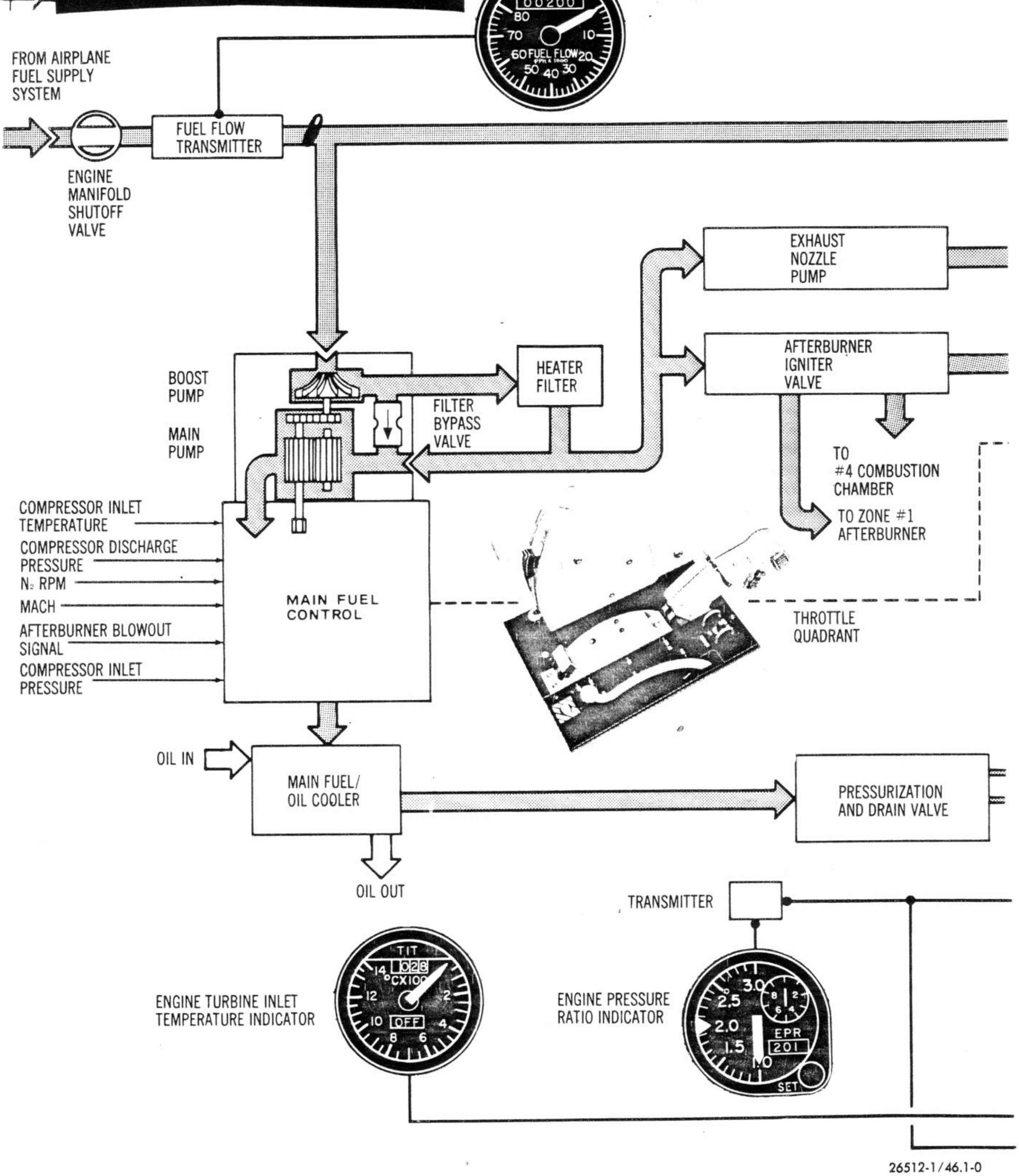


Figure 1-9 (Sheet 1)

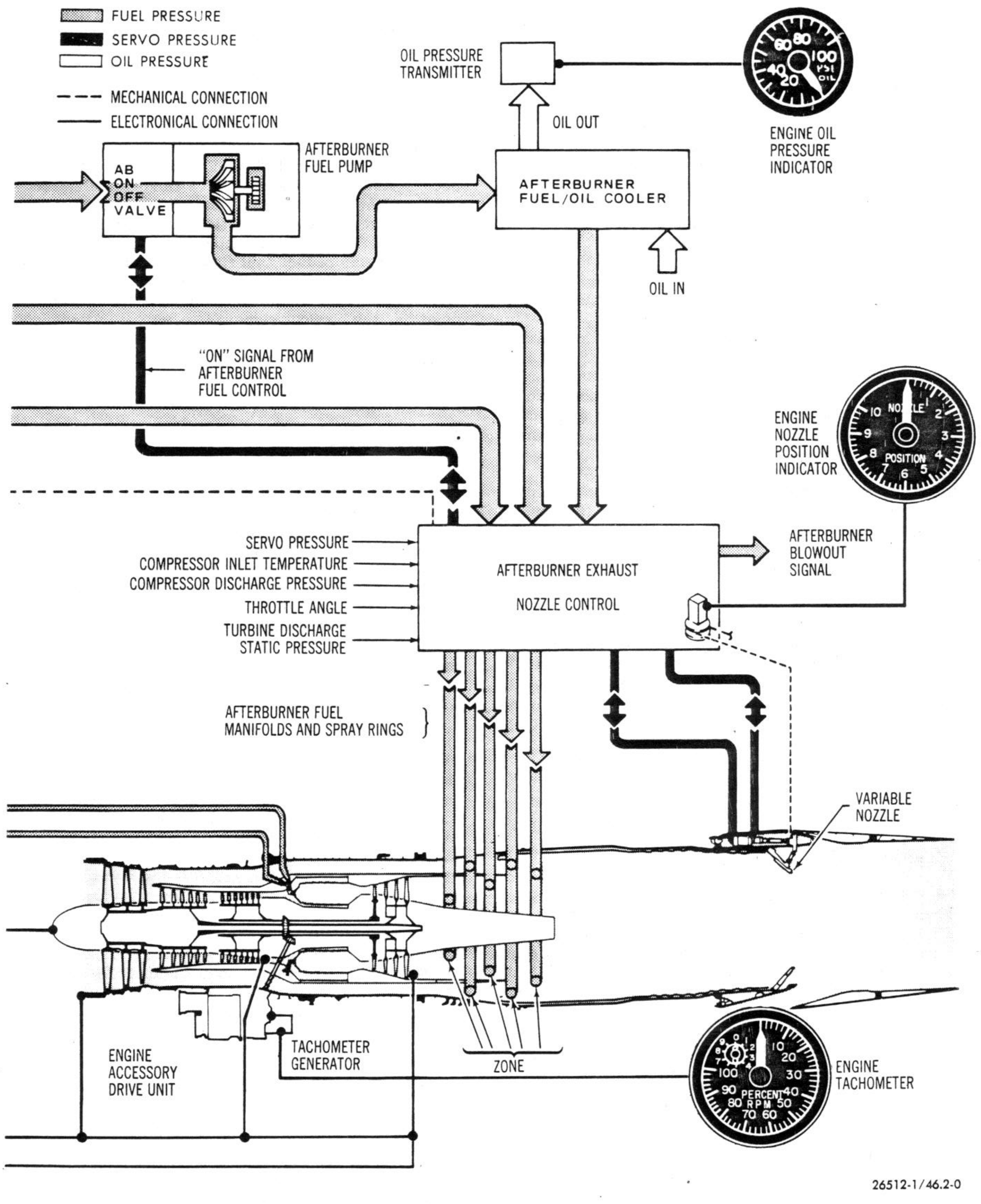


Figure 1-9 (Sheet 2)

the afterburner range, the afterburner fuel control pressurizes the afterburner first fuel manifold (zone 1), schedules light-off flow, and activates the variable nozzle system. This system senses a light-off pressure change and controls the exit area of the afterburner exhaust nozzle. Six free-floating, blow-in doors are located near the aft end of the afterburner. These doors open any time outside air pressure is greater than pressure inside the duct, allowing outside air to enter, thus maintaining nozzle efficiency. The trailing edge of the afterburner consists of a fixed cylindrical shroud.

AFTERBURNER FUEL SUPPLY

The afterburner fuel supply consists of the following major components: an exhaust nozzle pump, an afterburner fuel pump, an afterburner fuel control unit with integral exhaust nozzle control, and fuel spray rings. Fuel from the tanks flows through the flowmeter to the afterburner fuel pump. The exhaust nozzle pump is supplied fuel from the boost stage of the engine main fuel pump. The exhaust nozzle pump supplies fuel to the afterburner fuel control until a predetermined fuel flow rate is exceeded. At this flow rate, the afterburner fuel pump inlet is opened and begins to supply fuel to the afterburner fuel control unit. Fuel from the afterburner pump passes through a fuel-oil cooler before entering the afterburner fuel control unit. This unit includes a computer and a high-pressure flow section. Fuel is then directed to the spray rings where it is atomized and ignited in the afterburner combustion chamber. Five zones of afterburning can be selected through the afterburner fuel control unit, which schedules fuel to the spray rings in the various zones of the afterburner to serve as a function of throttle setting. When the throttle is advanced for afterburner initiation and high-pressure compressor speed exceeds approximately 80 percent rpm, the afterburner initiation valve schedules light-off fuel flow until afterburner light-off occurs, as sensed by the exhaust nozzle control.

AFTERBURNER IGNITION

The function of the afterburner ignition system is to provide a means of igniting fuel in the afterburner combustion chamber. When the system is actuated, fuel from the afterburner fuel system is injected into the aft end of No. 4 engine combustion chamber, thereby creating an excessively rich fuel-air mixture locally. This mixture results in a longer flame, which burns past the turbine stages to provide hot streak ignition for a second injection of fuel into the engine in the vicinity of the turbine exhaust section. This second hot streak continues aft and ignites the fuel that is sprayed into the afterburner combustion chamber.

VARIABLE EXHAUST NOZZLES

The variable nozzle system incrementally opens and closes the primary engine exhaust nozzle during afterburner modulation. The control is a hydro-mechanical computing device that determines and sets the nozzle area required to maintain a desired turbine pressure ratio during afterburner operation. The nozzle position is scheduled by the throttle setting and governed by turbine pressure ratio. The nozzle is closed for all ranges of non-afterburner operation, except for engine ground idle at which time it is positioned fully open for minimum thrust. The nozzle closes after the throttle is advanced 3 degrees above ground idle. If afterburner blowout occurs, the blowout signal valve is actuated and the nozzle closes. In addition, the afterburner fuel selector valve closes off fuel flow to all afterburner zones, and a signal is directed to the engine main fuel control to reduce fuel flow to the engine and open the engine bleed valves to prevent main engine overspeed. When the nozzle has moved to the closed position, the blowout signal is removed. Afterburner operation can again be initiated; however, the throttles must first be moved to a non-afterburning position in order to reset the afterburner control and ignition system.

IGNITION SYSTEM

The functions of the engine ignition system are to provide a means of initiating combustion in the combustion chambers during the starting cycle and to furnish an engine ignition source in the event of a flameout. Each engine has a dual main ignition system including two ignition exciters, two igniter plugs, an ignition alternator, and an automatic restart switch. The alternator is engine driven and is capable of providing sufficient energy to both exciters of the ignition system for ground starting or for air starts during flight conditions. Ignition alternator voltage is stepped up by transformer and capacitor circuits within the exciters to provide ionizing voltage for the igniter plugs. The alternator incorporates two independent, current-generating circuits for increased reliability. Should a combustion chamber flameout occur, the automatic restart circuit senses the rate of change of burner pressure and energizes the ignition system. Engine ignition is accomplished by two igniters in the lower combustion chambers of the engine. Advancing the throttle more than 3 degrees from OFF position actuates the throttle ignition switch for the engine and this action provides ignition. Electrical ignition is cut off when the ground start switch returns to OFF. This normally occurs when the starter centrifugal cutout switch opens at approximately 40 percent engine rpm. Ignition is also cut off when the throttle is retarded to less than approximately 3 degrees from OFF position.

STARTER SYSTEM

Each engine is provided with a pneumatic starter which is activated by air obtained from an external pneumatic ground starter cart, or by routing bleed air from the other engine (if operating). The pneumatic starter is composed of a turbine, gear train, over-running clutch with a speed-sensing device, and an overspeed disengagement mechanism with shear pin. When compressed air is used to start the engine, placing the ground start switch to the left or right and lifting the throttle out of the OFF position

opens the starter pressure shutoff valve and allows compressed air to operate the starter. When starting speed is reached, a centrifugal switch breaks the starter control circuit, allowing the control valve to close and shut off starter air.

ENGINE CONTROLS AND INDICATORS

THROTTLES

Two throttles on the throttle panel (figure 1-10) on the left console provide thrust setting adjustment for the engines. Throttle friction for both throttles is controlled by means of the friction lever located to the right of the throttles. Moving the lever toward INCR increases throttle friction, and moving the lever toward DECR decreases the friction. The throttles have detents for OFF, IDLE, MIL, and MAX AB, respectively. The throttles must be raised to go into or from the OFF position. The throttles must be raised to go forward of the MIL detent into the afterburner range. However, when retarding the throttles out of the afterburner range the throttles cam over the MIL detent without the need to raise the throttles. When the throttles are lifted to move them out of the OFF position, the throttle starter switches are actuated. Movement of the throttle past approximately 3 degrees forward of OFF activates the engine ignition system and fuel system. The right throttle includes a microphone switch and a speed brake switch. The left throttle includes an external light master switch.

GROUND START SWITCH

The engine ground start switch on the left console (figure 1-5) has three positions marked LEFT, OFF, and RIGHT. The switch is spring-loaded to OFF; however, when the switch holding coil is energized, the ground start switch remains in the position selected until the coil is deenergized. The holding relay of the switch is deenergized when, during starting, the centrifugal cutout switch in the starter opens the relay control circuit. This occurs when high-pressure compressor speed reaches approximately 40% rpm. If the ground start switch locks in the OFF position, the toggle must be lifted to reposition it. When the switch is placed in LEFT or RIGHT, 28-volt DC electrical power is supplied to arm the throttle starter switches. Power is also directed to energize the engine start relays and the ground start switch holding relay. Two throttle starter switches, one for each engine throttle, are actuated by throttle movement. When a throttle is lifted or moved out of OFF, the starter switch directs electrical power through the engine start relay to open the starter air shutoff for the respective engine if the ground start switch is in the applicable LEFT or RIGHT position. When the ground start switch is in the OFF position, electrical power is isolated from the engine starter system. Engine start counters in the left forward equipment bay record the number of starts.

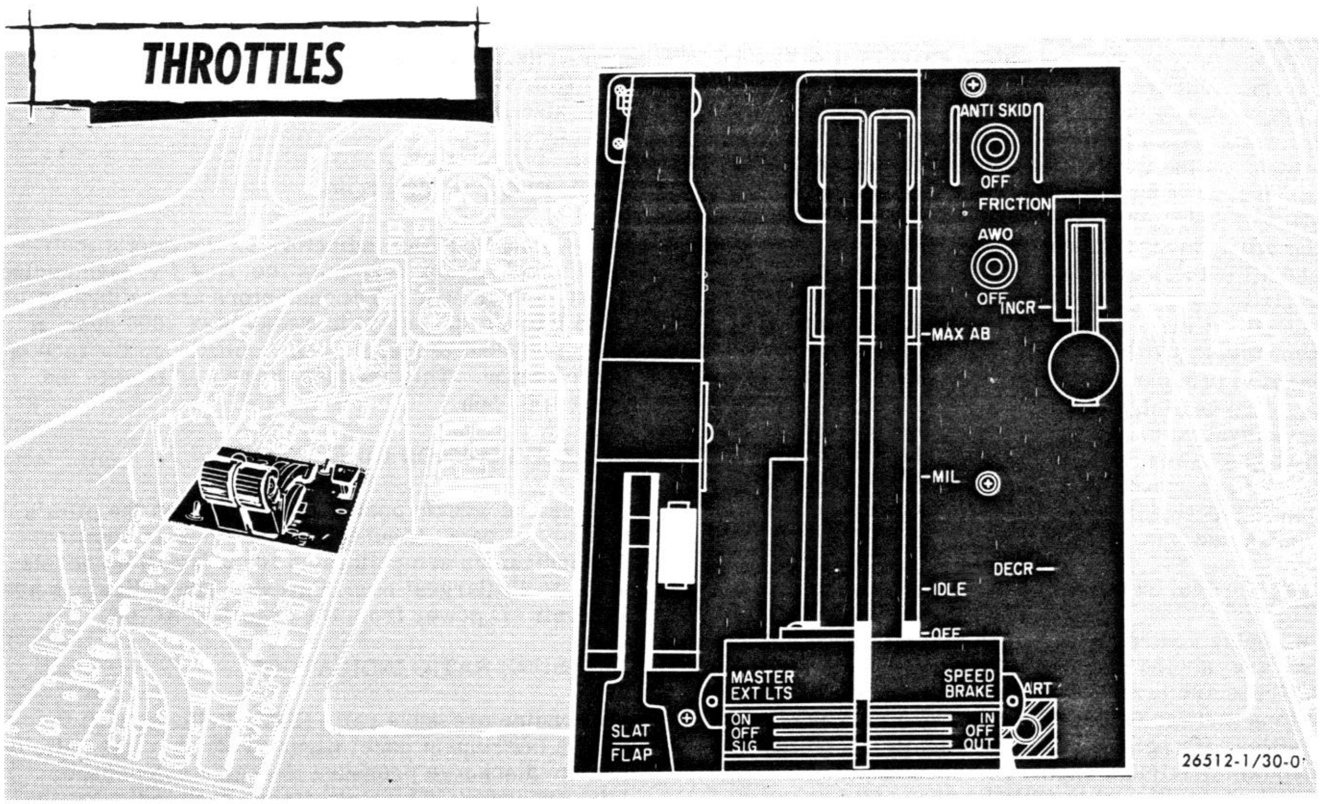


Figure 1-10

AIRSTART BUTTON

A pushbutton switch on the left console (figure 1-5) marked AIRSTART is for selecting air start ignition. When the airstart button is depressed, the airstart timer relay actuates and allows ignition generator power to energize the ignition exciters for both engines. The circuit will remain energized for approximately 55 seconds after the airstart button is released.

IGNITION CUTOFF SWITCH

The ignition cutoff switch on the ground check panel (figure 1-23) is labeled GRD IGNITION and has two positions marked NORM and OFF. When the switch is positioned OFF, a relay is energized which deactivates the engine electrical ignition system for both engines by grounding the ignition alternator output. When the switch is positioned NORM, the relay is deactivated and the ignition circuits are not grounded through this relay.

INLET SPIKES

Engine inlet air velocity is regulated throughout the entire aircraft speed range in order to maintain maximum engine performance. This regulation of the air inlet velocity is accomplished by a movable spike in the inlet of each engine. Each spike is a quarter circle, conical-shaped, variable diameter body that is independently movable forward and aft. The spikes are located in each air intake at the intersection of the wing lower surface and the fuselage boundary plate. Position and shape of the spikes are changed automatically to vary the inlet geometry and to control the inlet shock wave system. Local air pressure changes due to variations in inlet local mach and diffuser exit mach number are measured by mach wedges on the forward fuselage cheeks and by sensors in the spike control unit. These signals control operation of hydraulic actuators, which are powered by the utility hydraulic system to position the spike fore and aft (extend or retract) and adjust the spike cone angle by contracting and expanding the spike as required. If the system malfunctions, a pneumatic override controlled by switches in the crew compartment is provided to position and lock the spikes full forward and fully contracted. An electronic anti-icing system prevents ice formation on the sensors. Two (right and left) spike caution lamps illuminate when the aircraft mach number is less than 0.3 and the respective spike is not fully extended and fully contracted.

Spike Control Switches

Two spike control switches on the left console (figure 1-5) are labeled SPIKES, L and R respectively. The switches are lever-lock-type switches with two positions marked ORIDE and NORM. In the NORM position, the spikes are automatically controlled to maintain maximum engine performance. When either switch is positioned to ORIDE, pneumatic pressure is applied to the spike actuator to move the spike to the full forward and fully contracted position. The switch must be pulled out of the lock before it can be moved from either position.

Spike Test Buttons

Two spike test buttons on the ground check panel (figure 1-23) are for checking the operation of the spikes. The buttons are marked R SPIKE and L SPIKE. Depressing and holding either button will cause the respective spike to move to the full aft, fully expanded position. The spike caution lamps will illuminate while the spikes are in transit. When the buttons are released, the spikes will move to the full forward, fully contracted position.

Spike Caution Lamps

Two amber spike caution lamps, one for the spike in each engine inlet, are on the main caution lamp panel (figure 1-28). When illuminated, the letters L ENG SPIKE and R ENG SPIKE, respectively, are visible. A spike caution lamp illuminates when the aircraft mach number is less than 0.3 and the respective spike is not full forward and fully contracted. When the spike control switch is placed to ORIDE, the spike caution lamp will illuminate and remain illuminated until the spike has reached the full forward and fully contracted position. During spike self-test the lamps will illuminate until the spike has reached its full aft and full expanded position. The lamps operate on 28-volt DC electrical power.

TACHOMETERS

Two engine tachometers on the pilot's instrument panel (figure 1-3) indicate the percent of rpm of the high-pressure compressor (N2) in each engine. Each tachometer main dial is graduated from 0 to 100 percent rpm in increments of 2 percent; the subdial is graduated from 0 to 10 percent in increments of 1 percent.

FUEL FLOW INDICATORS

Two engine fuel flow indicators on the pilot's instrument panel (figure 1-3) show fuel flow for each engine in pounds per hour. The indicators are calibrated from 0 to 80,000 pph in increments of 2000 pph. A digital readout of fuel flow is displayed on the face of the indicator. This readout shows fuel flow to the nearest 50 pph.

NOZZLE POSITION INDICATORS

Two engine nozzle position indicators on the pilot's instrument panel (figure 1-3) show nozzle position. The indicators are calibrated for 0 (smallest nozzle area) to 10 (largest nozzle area). The indicators use 115-volt AC power from the essential AC bus.

PRESSURE RATIO INDICATORS

Two engine pressure ratio (EPR) indicators on the pilot's instrument panel (figure 1-3) show the ratio of turbine discharge pressure to engine compressor inlet pressure. The main dial of each indicator is calibrated from 1.0 to 3.0 in 0.1 increments. A smaller circular dial (sub-dial) on the indicator face is calibrated in 0.01 increments for precise reading. A set button on the lower right of each indicator

permits movement of a reference pointer on the perimeter of the indicator to serve as an index for computed EPR. The precise EPR position of the reference pointer is displayed by a digital readout window on the indicator face. The indicators are supplied 115-volt AC power from the essential AC bus.

TURBINE INLET TEMPERATURE INDICATORS

Two engine turbine inlet temperature (TIT) indicators on the pilot's instrument panel (figure 1-3) show turbine inlet temperature in degrees centigrade. The indicator dials are graduated from 0 to 1400 degrees in 50-degree increments. In addition, a digital readout of turbine inlet temperature in one degree increments is displayed. During engine start, a red flag with the letters HOT is displayed on the face of the indicator if TIT for the respective engine exceeds 705°C. Power to the TIT indicators is supplied from the 28-volt DC engine start bus. A flag marked OFF appears on the face of the indicator when power to the indicator is interrupted.

BLOW-IN DOORS

During low speed flight and ground operation, an additional amount of air is required for optimum engine performance. This additional air is provided by blow-in doors located forward on each inlet duct. The blow-in doors are positioned aerodynamically.

VORTEX DESTROYERS

The ingestion of foreign objects into the engine is prevented by an aerodynamic screen of engine bleed air that is directed down and outboard beneath each inlet through vortex destroyer air jets. The vortex destroyers serve to prevent the formation of vortexes below the inlet, thereby preventing foreign objects from being entrained in a vortex and sucked into the engine. When the weight of the aircraft is on the landing gear, a weight on wheels switch automatically activates the vortex destroyer air screen.

OVERSPEED LAMP

Two engine overspeed lamps, one for each engine, are on the main caution lamp panel (figure 1-28). When illuminated, the letters L ENG OVERSPEED and R ENG OVERSPEED, respectively, are visible. Illumination of either lamp indicates that its respective engine low pressure (N_1) is overspeeding.

ENGINE FIRE DETECTION SYSTEM

Engine fire detection is provided by sensing elements routed throughout each engine compartment. Should a fire or overheat condition occur, the rise in temperature is detected by the sensors which illuminate the respective left or right engine fire warning lamp. Shutoff valves isolate fuel and hydraulic fluid from the affected engine. Self-test features are incorporated in the system for maintenance checks and troubleshooting. A momentary contact toggle switch is provided for this purpose and is the center immediately below the FIRE PULL handles.

FIRE PULL HANDLES

Two fire pull handles, one for each engine, are on the left side, near the top of the Missile Control Officer's instrument panel (figure 1-4). They are positioned, respectively, for the left and right engine. Pulling either handle will shut off fuel and hydraulic fluid to the respective engine compartment. An engine fire warning lamp is in each handle and will illuminate whenever an overheat or a fire condition exists.

FIRE DETECTION GROUND TEST SWITCHES

Two fire detection ground test switches and a SHORT TEST button on the ground check panel (figure 1-23) are labeled R ENG and L ENG. The switches have three positions marked CONTROL BOX, NORM and ELEMENT. The switches are spring-loaded to the NORM (center) position and are used with the SHORT TEST button to ground check the system circuitry during maintenance or troubleshooting.

OIL SUPPLY SYSTEM

Each engine is equipped with an oil supply system which consists of an oil tank, a main supply pump, six scavenge pumps, a deoiler, two filters, an overboard breather pressurizing valve, a pressure valve, and three oil coolers (air-oil, fuel-oil, and afterburner fuel-oil). The air-oil cooler operates with engine bleed air. Oil is fed to the main oil supply pump from the oil tank. It is then pumped in series through the two filters, the air-oil cooler, fuel-oil cooler, and afterburner fuel-oil cooler. Oil flow through the fuel-oil coolers is controlled by temperature and pressure-sensing bypass valves. The oil is then directed to the engine bearings and to the accessory gearbox. Scavenge pumps return the oil to the oil tank. Capacity of the tank is five gallons, four gallons of which are usable.

OIL QUANTITY INDICATOR

The engine oil quantity indicator on the aft end of the left console (figure 1-5) is a dual-indicating instrument with two displays labeled L and R for the left and right engine, respectively. Each display is graduated in one quart increments from 0 to 16. A pointer for each display provides an indication of the number of quarts of oil remaining in each oil tank.

Oil Quantity Indicator Test Button

The engine oil quantity indicator test button beside the oil quantity indicator provides a means of checking the indicator. When the button is depressed and held, the pointers will drive to predetermined values of 5 quarts on the left display and 5.7 quarts on the right display. When the button is released, the pointers will return to their previous indications.

Oil Low Caution Lamp

An oil low caution lamp on the main caution lamp panel (figure 1-28) illuminates any time the oil level in either the left or right engine oil supply tank drops to four quarts. When the lamp is illuminated, the letters OIL LOW are visible.

Oil Hot Caution Lamps

The two engine oil hot caution lamps are on the main caution lamp panel (figure 1-28). When the oil temperature of either engine exceeds 225°F, the associated lamp will illuminate. When illuminated, the following letters will be visible in the lens of the respective lamp: L ENG OIL HOT; and R ENG OIL HOT.

Oil Pressure Indicators

Two engine oil pressure indicators on the pilot's instrument panel (figure 1-3) indicate engine oil pressure in pounds per square inch. The indicators are calibrated from 0 to 100 psi in increments of 5 psi. The oil pressure indicating system operates on 26-volt AC power that is supplied from the essential AC bus through a transformer.

FUEL SUPPLY SYSTEM

The fuel system (figure 1-11) consists of a forward and aft integral fuselage tank, two integral wing tanks, an integral vent tank, and the associated fuel pumps, controls, and indicators. During normal operation, the left engine receives fuel from the forward fuselage tank, and the right engine receives fuel from the aft fuselage tank. Fuel from the wing tanks is transferred to the fuselage tanks before being delivered to the engines. The fuel system employs ten fuel pumps, six of which deliver fuel to the engines and four are used to transfer fuel from the wing tanks to the fuselage tanks. Single-point refueling is provided for ground servicing and is accomplished through a standard ground refueling receptacle on the left side of the fuselage. All tanks are equipped with refueling automatic shutoff valves.

FUEL TANKS

The fuselage tanks are two separate tanks, identified as the forward and aft tanks. The forward tanks extend from the aft bulkhead of the equipment bay to the bulkhead forward of the main wheel well. The forward tank is divided into three separate bays. The bays are identified as F-1, F-2, and the trap tank (reservoir tank). The F bays and trap tank are interconnected by standpipes and one-way flow flapper valves. The flapper valves allow fuel to flow from the F bays to the trap tank. The portion of the forward tank consisting of the wing box carry-through structure and the fuselage below the box and behind the weapons bay is called the trap tank. The trap tank is in two sections, a lower and an upper tank. The trap tank serves as a fuel reservoir and retains approximately 2550 pounds of fuel after all other fuel has been used. When the usable fuel level in the forward tank drops below approximately 5000 pounds, the FUEL LOW caution light is illuminated. The aft fuel tank extends from aft of the main gear wheel well to the bulkhead at the rear of the fuselage structure. The aft tank is divided into two bays, A-1 and A-2. Each wing has an integral fuel tank that extends from the wing pivot structure to nearly the wing tips. Fuel in the wing tanks cannot be fed directly to the engines but must first be transferred to the fuselage tanks. A vent tank located in the vertical fin is provided for fuel expansion and for venting the fuselage and wing tanks. See figure 1-12 for location of fuel tanks and fuel quantity.

FUEL PUMPS

There are ten fuel pumps in the fuel system that operate on 115 volt, three-phase, 400 cycle AC power. The six fuselage fuel pumps are dual inlet booster pumps, and the four wing fuel pumps are single inlet transfer pumps. Boost pumps 1 and 3 are in bay F-2; 2 and 4 are in the trap tank; 5 and 6 are in bay A-1. Transfer pumps 7 and 9 are in the left wing, and transfer pumps 8 and 10 are in the right wing. Pumps 3, 4, 5, and 6 are the primary engine feed pumps; 1 and 2 are standby engine feed pumps. Number 1 boost pump is a standby pump and operates continuously with the engine feed selector switch in any position except OFF. When not needed for engine fuel supply, the fuel provided by pump 1 is circulated into the trap tank through a pressure relief valve. The number 2 standby pump is energized whenever any one of the manifold pressure-sensing switches sense a manifold pressure less than 16.1 psi above tank pressure.

FUEL TRANSFER

In order to use the fuel in the wing tanks, it must be transferred to the fuselage tanks. The forward and aft refueling valves will open during transfer operation any time the tank levels have dropped 400 pounds in the forward tank and/or 250 pounds in the aft tank. Refueling valves cannot be controlled by the pilot. The activation of the fuel transfer system is controlled by the fuel transfer switch on the fuel panel. Fuel level in the fuselage tanks is maintained by float valves which permit the refueling valves to open and cause the refueling valves to close when the tanks are full. When transferring fuel, the fuel pump low-pressure advisory lamps should be used in conjunction with the fuel quantity indicator to determine when a particular tank is empty. When emptying the wing tanks, the wing transfer fuel pump low-pressure lamps may not illuminate simultaneously, depending on the sweep angle. With the wings swept forward, the outboard transfer pumps will run out of fuel before the inboard pumps. With the wings swept aft, the reverse will occur.

FUEL PRESSURIZATION AND VENT SYSTEM

The fuel pressurization system air is obtained from the engine compressor bleed line and is used to provide pressure for the fuel tanks. The system maintains a pressure between 5.0 and 6.0 psig in the tanks by means of the fuel tank vent and pressurization control valve. Should the fuel tank pressure approach 6.0 psig, the vent valve opens to vent the excess air overboard through the vent/dump outlet at the lower aft end of the fuselage. If the pressure decreases, the valve controls air into the tank to maintain pressure.

FUEL DUMP

The aircraft is capable of dumping fuel at a minimum rate of 2300 pounds per minute. During dumping operation, all fuel is automatically transferred to the

forward fuselage tank. Fuel tank pressure then forces the fuel from the forward fuselage tank overboard through the fuel dump line. The fuel dump outlet is located between and aft of the engines directly beneath the rudder. All fuel except that in the trap tank can be dumped.

Note

If dumping operation is necessary during afterburner operation, the fuel may be ignited by the exhaust flame. This should cause no concern because the fire will remain behind the aircraft. Any nearby aircraft should be advised to stay well clear while fuel dumping is in progress.

To eliminate prolonged fuel dripping from the fuel dump outlet after dumping is discontinued, the fuel system may be momentarily depressurized to clear residual fuel from the fuel dump lines. This will occur automatically when the landing gear is extended for landing. During fuel dumping operations, it should be noted that the automatic center-of-gravity control will not operate normally.

SINGLE-POINT REFUELING

The aircraft is equipped with a single-point refueling system which enables all aircraft fuel tanks to be pressure-filled simultaneously from a single refueling receptacle. During ground refueling operations, fuel flows through the refueling receptacle and refueling manifold into the fuel tanks. As each tank fills, a float-operated valve automatically closes the refuel valve, stopping flow to the tank. The single-point refueling receptacle is on the left side of the fuselage forward of the engine air intake.

ENGINE FUEL FEED

The engine fuel feed selector controls the sequence of fuel flow to the engines and the transfer of fuel between the fuel tanks. The engine fuel feed system, when functioning in the automatic mode, maintains a predetermined fuel quantity difference between the forward fuel tank and the aft fuel tank in order to control the aircraft center of gravity. There are four modes of engine feed fuel flow controlled by the engine feed selector switch on the fuel control panel (figure 1-13), immediately forward of the throttles. The first mode is OFF and when the selector is in this position, all of the fuel pumps are shut off. The second mode is FWD and in this position the forward tank will feed both engines. The third mode is AUTO which is the normal position for fuel transfer to the engines. In this mode the forward tanks feed the left engine fuel manifold and the aft tanks feed the right engine fuel manifold. A 5600 (±250) pound differential is maintained between the forward fuselage tank, with the 5600 pounds more maintained in the forward tank. This differential is maintained by automatic transfer of fuel from the aft tank to the forward tank. If the differential is greater than 5850 (±100) pounds, the aft fuel pumps are shut off and fuel is used from the forward tank until the differential is re-established, at which point both forward and aft pumps again operate. When the differential is less than 5350 (±100) pounds, the transfer valve opens and aft fuel is pumped into the forward tank until the differential is re-established, at which time the transfer valve closes. For the fourth mode of operation, the engine feed selector switch is positioned to AFT and in this position the aft tank will feed both engines. In the fifth mode which is BOTH, the boost pump feed is the same as for the automatic mode except that no specific fuel differential is maintained between the forward and the aft tank.

FUEL CONTROL PANEL

All in-flight fuel system control switches and associated fuel pump low-pressure advisory lamps are on the fuel control panel on the left console (figure 1-13).

Engine Fuel Feed Selector

The engine fuel feed selector on the fuel control panel (figure 1-13) is a rotary, five-positioned detent selector placarded OFF, FWD, AUTO, AFT, and BOTH. When the selector is rotated to OFF, all fuel boost pumps are de-energized. When the selector is rotated to FWD, boost pumps 1, 3, and 4 are energized, and boost pump 2 is placed on standby. In this configuration, both engines are fed from the forward fuel tank. When the selector is rotated to AUTO, boost pumps 1, 3, 4, 5, and 6 are energized, and boost pump 2 is placed on standby. In this configuration, fuel is fed simultaneously from both the forward and aft fuselage fuel tanks with a differential of 5600 (±250) pounds automatically maintained between the two tanks. When the selector is rotated to AFT position, boost pumps 1, 5, and 6 are energized, boost pump 2 is on standby, and both engines are fed from the aft fuselage tank. When the selector is rotated to BOTH, boost pumps 1, 3, 4, 5, and 6 are energized, and boost pump 2 is on standby. In this configuration, the left engine is fed from the forward fuselage tank, and the right engine is fed from the aft fuselage tank.

Fuel Transfer Selector

The fuel transfer selector on the fuel control panel (figure 1-13) is a four-positioned, rotary selector labeled TRANSFER and marked OFF, WG, EXT, and AUTO. When the selector is rotated to WG, four transfer pumps, two in each wing tank, are energized and fuel is transferred from the wing tanks. The EXT position of the selector has no operational function at this time. The AUTO position functions the same as the WING position.

Fuel Pump Low-Pressure Advisory Lamps

Ten green lamps on the left side of the fuel control panel (figure 1-13) are fuel pump low-pressure advisory lamps. When a fuel pump is energized and is not generating the required minimum pressure (3.5 psi), the corresponding green lamp illuminates. The uppermost two lamps are for the standby pumps in the forward tank. The next two lamps are for the forward fuselage tank pumps. The next two lamps are for the aft fuselage tank pumps. The next two

FUEL SUPPLY SYSTEM

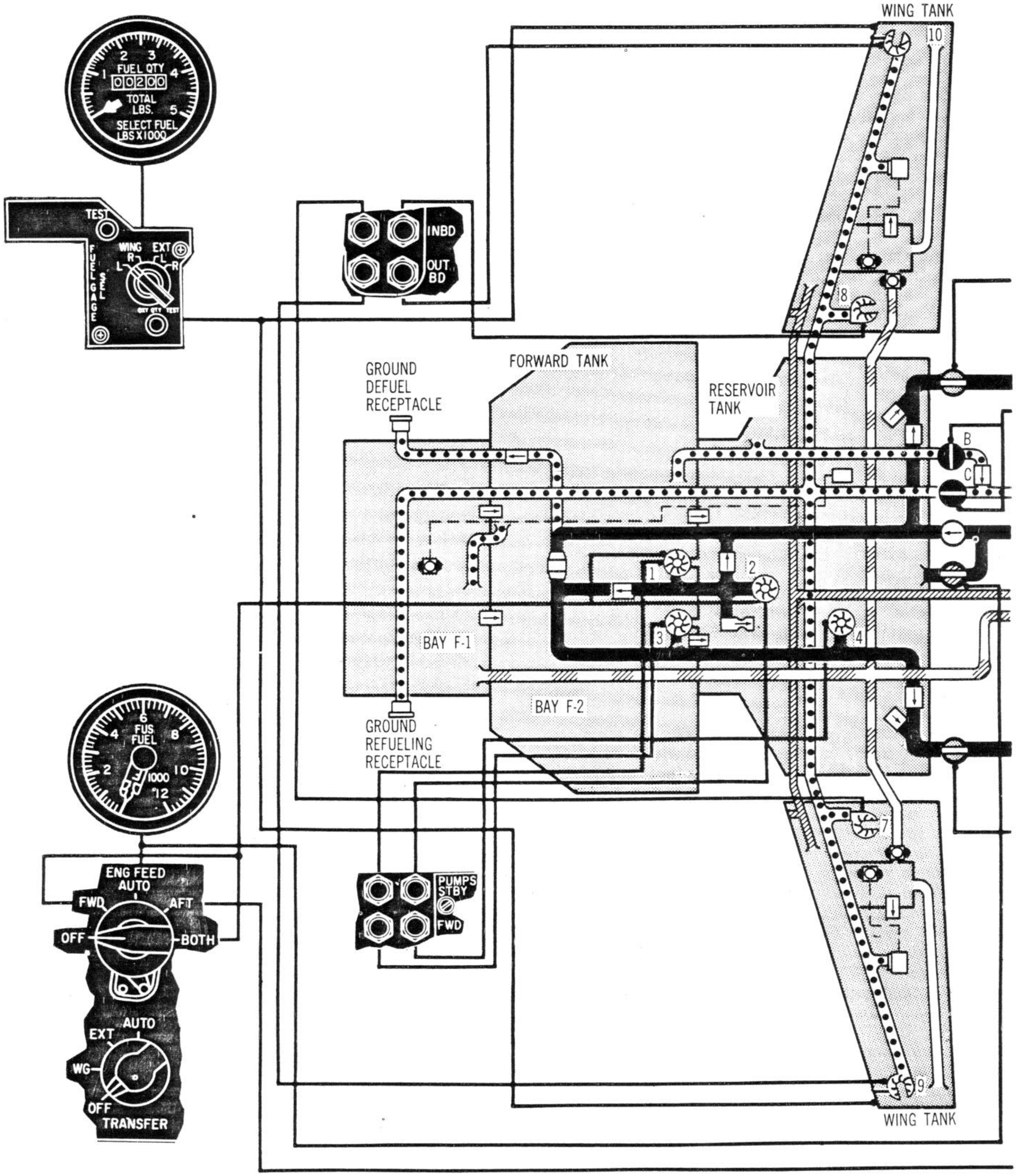


Figure 1-11 (Sheet 1)

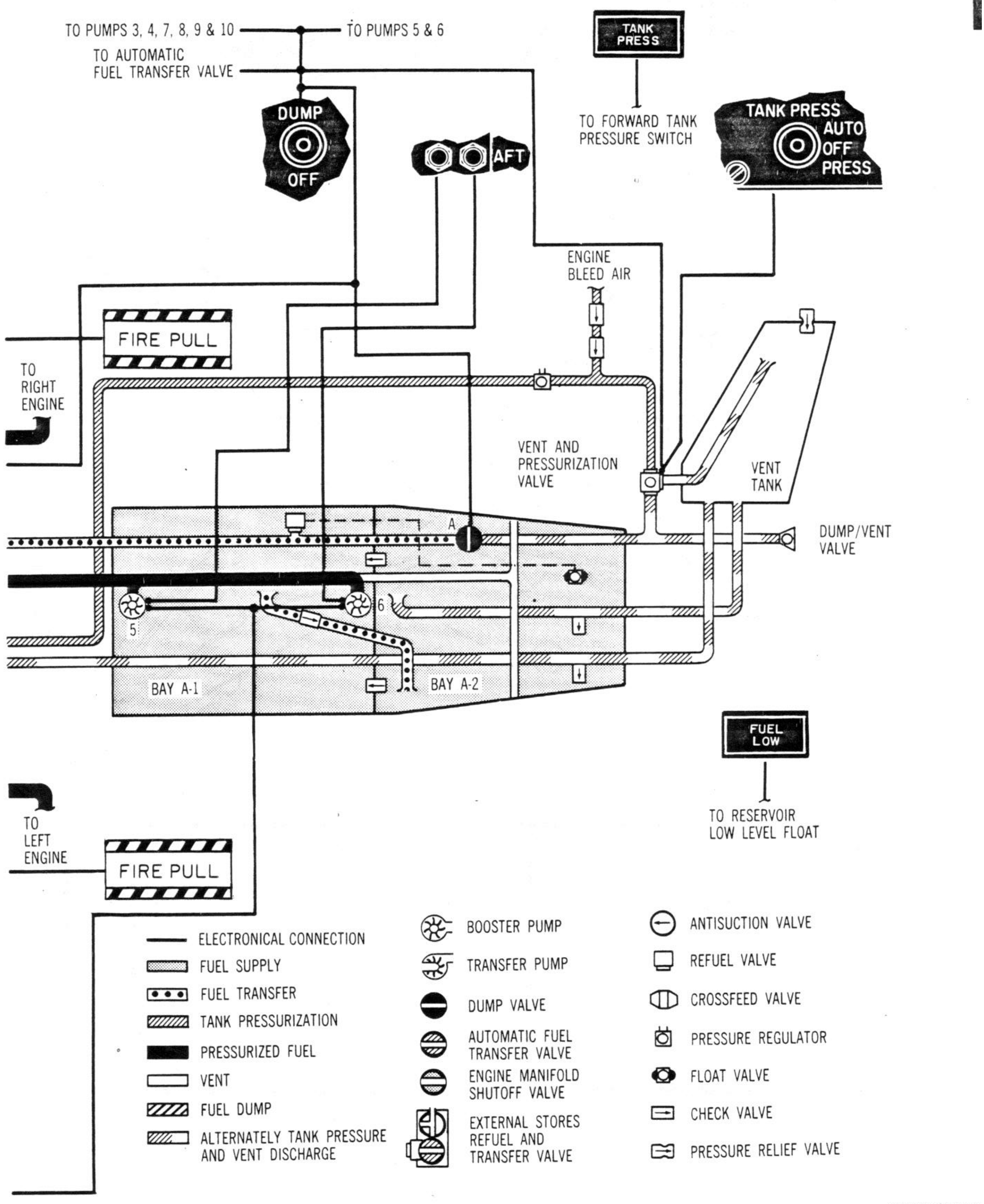
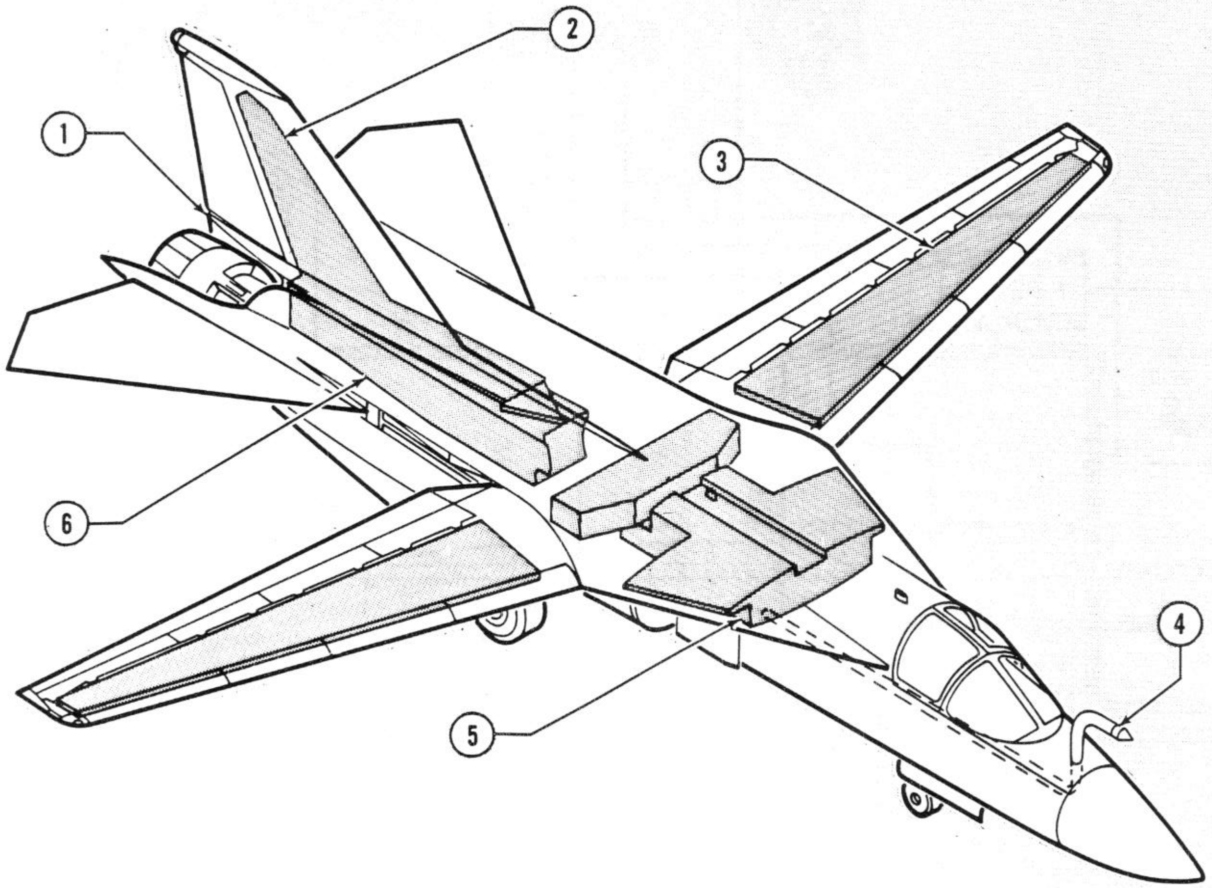


Figure 1-11 (Sheet 2)

FUEL QUANTITY

TYPICAL



1. VENT/DUMP OUTLET
2. VENT TANK
3. WING TANK
4. REFUELING PROBE
5. FORWARD FUSELAGE TANK
6. AFT FUSELAGE TANK

FUEL CAPACITIES (JP-5)

INTERNAL FUEL (USEABLE)	25,542 LBS
FORWARD FUSELAGE	12,995 LBS
AFT FUSELAGE	7,000 LBS
WINGS	5,245 LBS
LINES	302 LBS
UNUSEABLE FUEL	195 LBS

NOTE

WEIGHTS ARE BASED ON 6.8 POUNDS PER GALLON OF JP-5 FUEL, STANDARD DAY CONDITIONS

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Figure 1-12

FUEL CONTROL PANEL

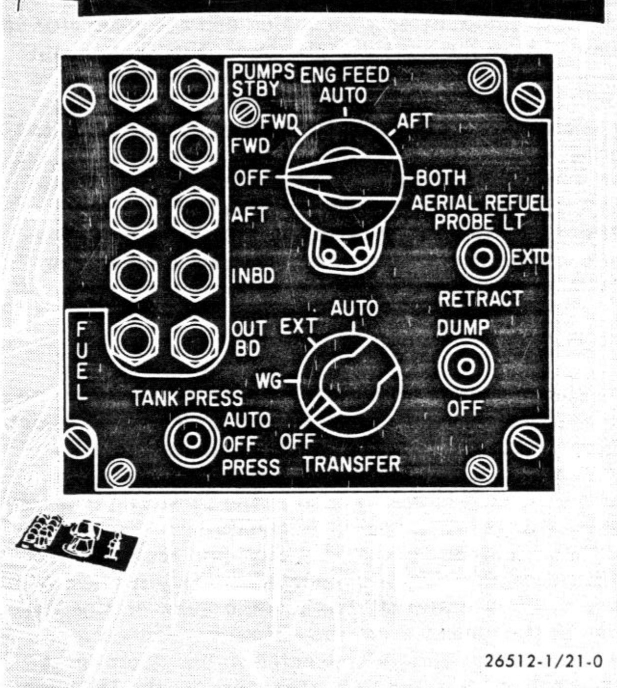

Figure 1-13

lamps are for the wing inboard transfer pumps, and the lower two lamps are for the wing outboard transfer pumps. The fuel pump low-pressure indicator lamps may not illuminate simultaneously.

Fuel Dump Switch

The dump switch on the fuel control panel (figure 1-13) is a two-positioned switch marked DUMP and OFF. When the switch is in the off position, dump valves A and B are closed and C is open. When the switch is in the dump position, dump valves A and B are opened, C is closed, the automatic transfer valve is opened, the tanks are pressurized and fuel booster pumps 5 and 6 and fuel transfer pumps 7, 8, 9, and 10 are energized, thereby transferring fuel from the wing and aft tanks to the forward tank to be dumped. The fuel tanks will pressurize when the dump switch is in the dump position regardless of the position of the fuel tank pressurization selector switch, or the landing gear.

Fuel Tank Pressurization Selector Switch

The fuel tank pressurization selector switch on the fuel control panel (figure 1-13) is a three-positioned, lever-lock toggle switch marked AUTO, OFF, and PRESS. When the switch is positioned to AUTO, the fuel tanks are pressurized, except when the landing gear is down. When the switch is placed to OFF, the pressurization airflow to the tanks is turned off and the tanks are vented. When the switch is placed to PRESS and pressurization air is available, fuel tank pressurization is maintained with the landing gear down. If the tanks are depressurized at low fuel levels, it will take considerably longer to repressurize than it does when the tanks are full. Therefore, the tank pressurization caution lamp will remain illuminated for a longer time.

FUEL SYSTEM CAUTION LAMPS AND INDICATORS

Fuel Tank Pressurization Caution Lamp

The amber fuel tank pressurization lamp on the main caution lamp panel (figure 1-28) illuminates when fuel tank air pressure drops below approximately 3.3 psi during flight with the landing gear retracted. The lamp also illuminates any time the fuel tank pressurization switch is in AUTO and the landing gear is extended. When the lamp illuminates, the letters TANK PRESS are visible.

Fuel Distribution Caution Lamp

The amber fuel distribution caution lamp on the main caution lamp panel (figure 1-28) illuminates when the fuel distribution between the forward and aft tanks is out of tolerance. The lamp is illuminated when the difference between the forward and aft tanks is less than 5000 (±150) pounds, or more than 6500 (±150) pounds. When illuminated, the letters FUEL DISTRIB are visible.

Fuel Manifold Low Fuel Pressure Caution Lamps

Two amber low fuel pressure caution lamps are on the main caution lamp panel (figure 1-28). The letters L FUEL PRESS or R FUEL PRESS are visible when the respective lamp is illuminated. The applicable lamp illuminates any time the fuel pressure in the right or left fuel manifold is less than 15.5 psi.

Fuel Low Caution Lamp

The amber fuel low caution lamp on the main caution lamp panel (figure 1-28) illuminates any time usable fuel in the forward tank is less than approximately 5000 pounds. When the lamp is illuminated, the letters FUEL LOW are visible.

Fuel Quantity Indicators

Two fuel quantity indicators are on the pilot's instrument panel (figure 1-3) immediately above the center console. One indicator, labeled FUS FUEL, with two pointers, one for the forward fuselage tank and one for the aft fuselage tank, displays fuselage fuel quantity. The second indicator has a pointer and a digital counter display. The pointer is a select indicator which displays the fuel quantity in the wing tanks, left or right. The digital readout continuously displays the total quantity of fuel in all tanks. The four-positioned, rotary fuel gage selector switch, with positions marked WING (L-R), EXT (L-R), permits selection of each tank for readout.

Fuel Quantity Indicator Test Button

The fuel quantity indicator test button on the pilot's instrument panel (figure 1-3) tests the fuselage fuel quantity and total/select fuel quantity indicator. When the test button is depressed, each of the three pointers and the total fuel digital counter will drive to the following indications: Forward and aft tank pointers, 2000 (±240); select tank pointer, 2000 (±100); total fuel digital counter, 2000 (±250). The indicators will either increase or decrease to the 2000-pound reading, depending on the quantity of fuel in the tanks. A

normal confidenee check of the fuel quantity indicators may be made by depressing the test button long enough to observe movement of the pointers and counter. When the button is released, the pointers and counter will return to their original readings.

Fuel Gage Selector Switch

The fuel gage selector switch on the pilot's instrument panel (figure 1-3) is a four-positioned rotary switch marked WING (L-R), EXT (L-R). Placing the switch to the desired tank will make it possible to read the quantity of fuel remaining in the respective tank on the select fuel quantity indicator.

Stores Refuel Power Switch

The stores refuel power switch on the ground check panel (figure 1-23) has no operational function at this time except that it should be positioned to NORM at all times. REFUEL position causes all other aircraft electrical systems to be de-energized.

ELECTRICAL POWER SUPPLY SYSTEM

The electrical power supply system provides 115/200 volt, three-phase, 400 cycle, AC power from two AC generators, driven by constant speed drive units (CSD), one mounted on each engine. Two separate transformer-rectifier units convert power from AC buses to provide 28-volt DC power. An emergency system, consisting of a hydraulic-motor-driven AC generator, is provided to supply the power loads essential to flight and for safe landing. Transfer to the emergency system is automatic when main generator power to the AC buses is lost.

AC PRIMARY SUPPLY

There are two, 8000 rpm, 67.5 KVA generators driven by CSD units, one for each engine, that supply the primary AC electrical power to the system AC buses (figure 1-14). Either generator is individually capable of supporting the normal electrical loads. CSD units regulate frequency at 400 cycles per second while the system is protected from under/over voltage, under/over frequency, and overcurrent conditions by generator control units, one for each generator. There are three AC buses, a left main, a right main, and an essential. Normally, the left generator feeds the left main bus, and the right generator feeds the right main and the essential buses. Power is carried from each generator to the respective buses by separate multiple wire feeders through power transfer contractors. These contractors provide an automatic bus-tie function should either generator fail. If a fault or malfunction occurs, resulting in under/over voltage, under/over frequency, or overcurrent, the associated generator control unit disconnects the generator from its respective bus. It will also de-excite the generator in cases of under/over voltage, and/or overcurrent. In all instances the bus concerned is automatically tied in with the other generator by the power transfer contactor. If the malfunction is corrected, the generator may be brought back into the system by cycling the generator from ON to OFF and back to ON. Should an excessive amount of heat occur in a CSD unit, a thermal device will automatically decouple the drive from the engine. There are no provisions for recoupling in flight.

EMERGENCY ELECTRICAL POWER

Emergency AC power is provided by a hydraulic-motor-driven 10 KVA generator system. If all power is lost to the primary bus system, the hydraulic motor is automatically turned on and the generator is connected to the essential AC bus. Utility hydraulic system pressure, controlled by a solenoid actuated hydraulic shutoff valve, drives the hydraulic motor. DC power from the main DC bus is normally applied to the valve to keep it closed and thus keep the motor from turning. Failure of main DC power will automatically start the emergency generator turning. Emergency DC power is supplied through the transformer-rectifier system to the essential and crew station DC buses.

DC POWER

DC electrical power (figure 1-15) is provided by two 28-volt DC transformer-rectifier (T/R's) units which energize three DC buses, a main, an essential, and a crew station. The essential bus consists of two interconnected buses located in the forward equipment bay and the crew module, respectively. The main DC bus receives power from the main transformer-rectifier unit which is connected to the left main AC bus. The essential DC bus and the crew station DC bus receive power from the essential transformer-rectifier unit which is connected to the essential AC bus. A bus-tie contactor interconnects the DC buses and the transformer-rectifiers supply the total DC load in parallel. During emergency AC generator operation, the main DC bus is dropped from the system.

ELECTRICAL CONTROL PANEL

The electrical control panel (figure 1-16) placarded ELEC on the center console contains all of the electrical system operating switches.

Emergency Generator Switch

The emergency generator switch on the electrical control panel (figure 1-16) has three positions placarded AUTO, ON, and TEST. Placing the switch in AUTO arms the emergency generator system. If all AC power is lost, the emergency generator is automatically turned on and connected to the essential AC bus to provide emergency AC power. When normal AC power is available, this generator is not connected or operating while the switch is in AUTO. In the ON and TEST positions, the generator is operating but will not be connected to the essential AC bus unless all AC power is lost. By splitting the main DC bus and the essential DC bus, the TEST position also provides a method for checking the two transformer-rectifiers supplying DC power. If both T/R's are operating, the electrical power flow indicator will display NORM or TIE. If the essential DC bus becomes de-energized due to an inoperative T/R, the total temperature indicator will display an OFF flag and the AC bus source indicator will show a cross-hatched surface. If the main DC bus becomes de-energized due to an inoperative T/R, the digital data indicator will display yellow and black flags. A lamp

AC ELECTRICAL POWER SUPPLY SYSTEM

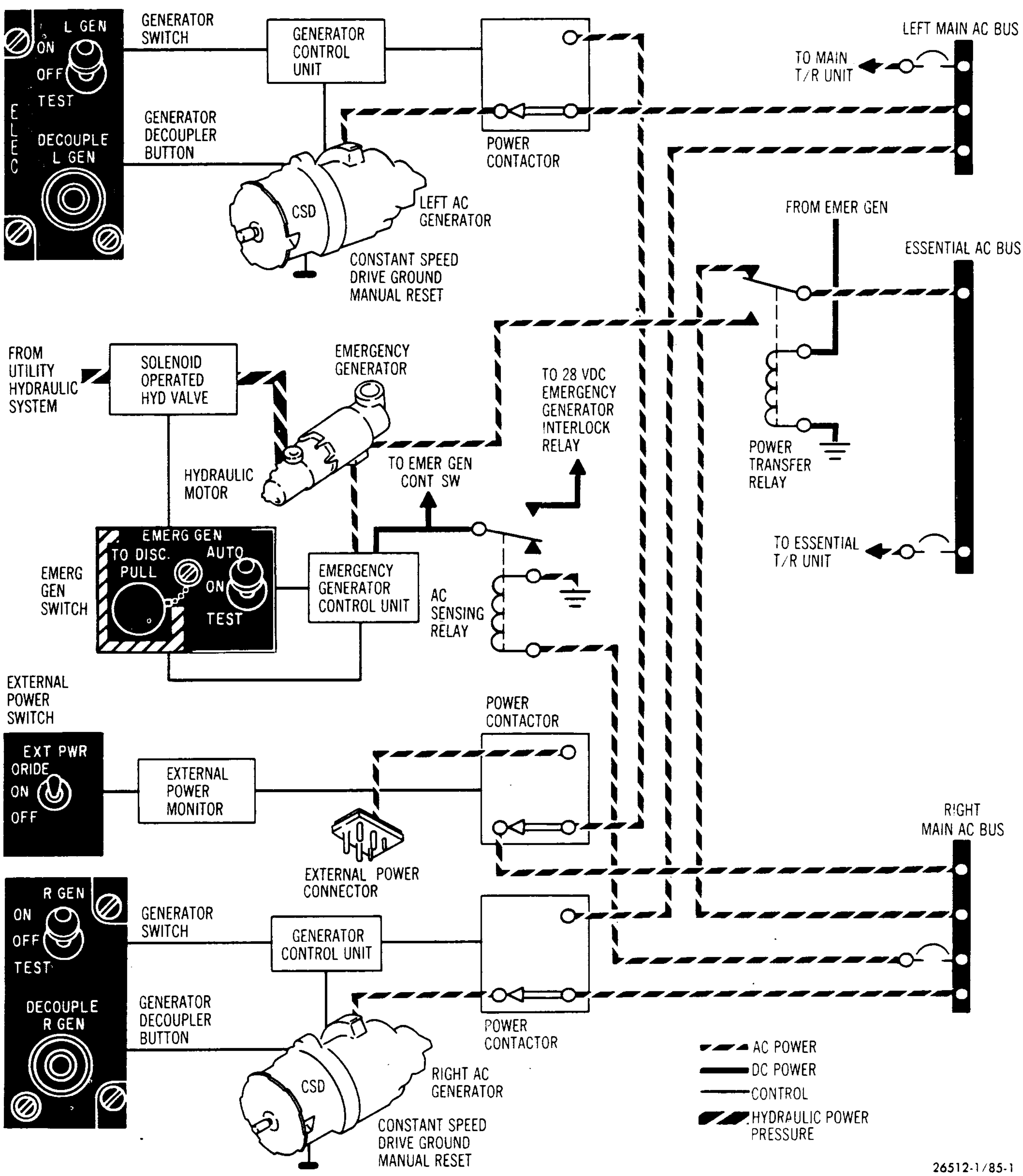

Figure 1-14

DC ELECTRICAL POWER SUPPLY SYSTEM

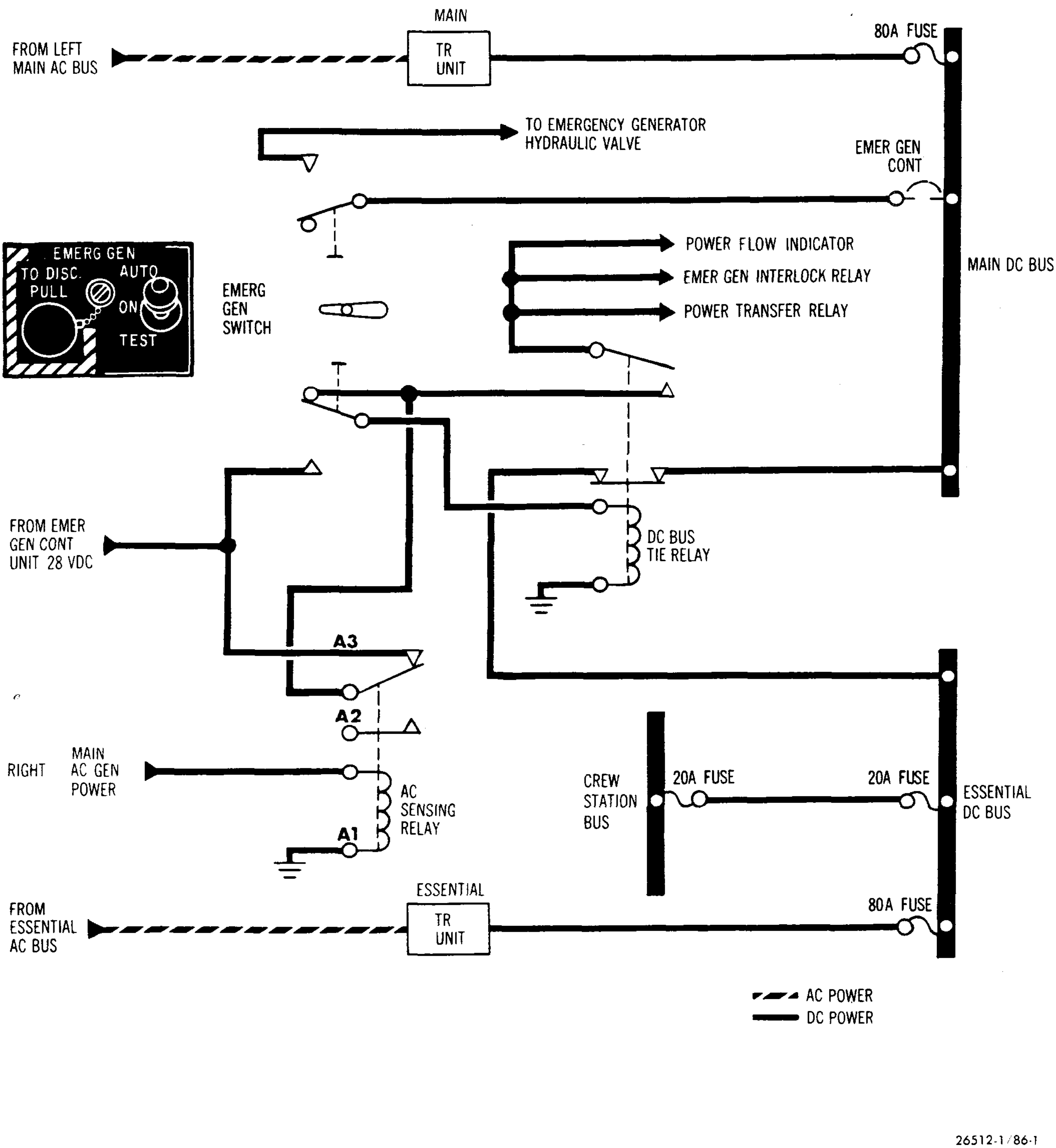

Figure 1-15

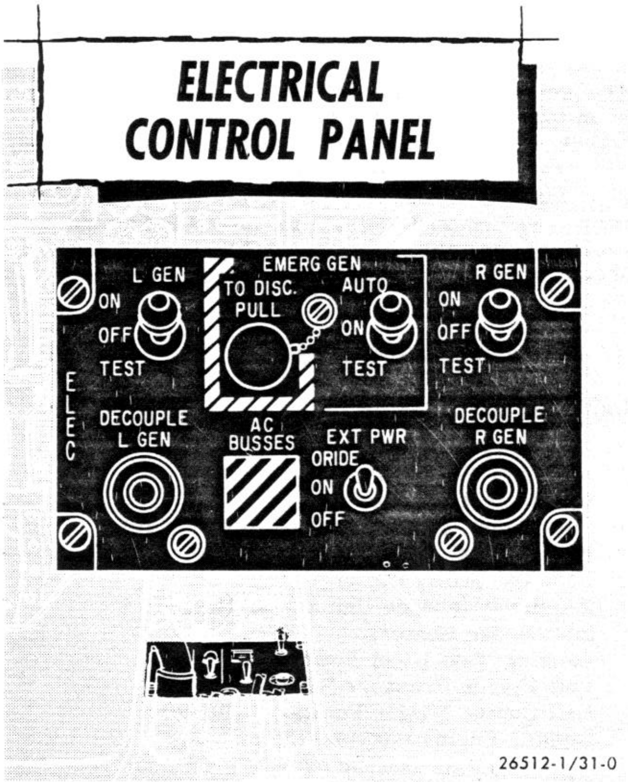

Figure 1-16

in the TO DISC PULL switch is illuminated, indicating operation of the emergency generator when the switch is positioned to either ON or TEST, and will also be illuminated any time emergency AC power is being supplied to the system while the switch is positioned to AUTO.

To Disconnect Pull Switch (TO DISC PULL)

The TO DISC PULL switch on the electrical control panel (figure 1-16) is provided for the purpose of positively turning off the emergency generator, if desired. When the switch is pulled out, the emergency generator is de-excited and will no longer generate AC power.

Generator Switches

The two generator switches on the electrical control panel (figure 1-16) placarded L GEN and R GEN are lever-lock-type toggle switches with positions marked OFF, ON, and TEST. In the OFF position, the generator is not excited, the power contactor is open, and the generator system is reset.

Note

If a generator is de-excited while connected to the bus, it will not automatically reset, even though the fault condition is cleared. The generator switch must be positioned to OFF to reset the system.

Positioning the switch to ON will excite the generator, close the power contactor, and connect it to its respective bus, if the power is within limits. In the TEST position, the generator will be excited, but disconnected from its bus. The TEST position can be used to check generator operation without connecting it to a bus.

Generator Decouple Pushbuttons

The generator decouple pushbuttons placarded DECOUPLE L GEN and DECOUPLE R GEN on the electrical control panel (figure 1-16) are to actuate the constant speed drive decoupler. When a pushbutton is depressed, a solenoid within the constant speed drive is actuated, causing the CSD gearing to be physically disconnected from the input shaft. Once decoupled, the CSD cannot be reconnected in flight.

CAUTION

The generator decouple switch must be depressed only momentarily (not more than 5 seconds).

Electrical Power Flow Indicator

The electrical power flow indicator on the electrical control panel (figure 1-16) is a flip-flop type indicator, placarded AC BUSSES, and displays the various bus configurations. If both buses are receiving power from their respective generator, the indicator will display NORM, indicating that the buses are isolated from each other and are operating normally. If only one generator is providing power for the AC buses, the indicator will display TIE. When the emergency generator is operating and supplying power to the essential AC bus, the indicator will display EMERG. When ground power is connected to the aircraft and supplying power to the AC buses, the indicator will display TIE. When there is no AC power being applied to the aircraft, the indicator will display a crosshatched surface.

External Power Switch

The external power switch on the electrical control panel is a toggle switch having positions marked OFF, ON, and ORIDE. In the OFF position, external power cannot be supplied to the AC buses. In the ON position with neither engine operating, external power supplies total aircraft power. With the left engine operating, the left main AC generator will supply total aircraft electrical load, and external power is disconnected from the AC buses. With only the right engine operating, the right main AC generator supplies power to the right main and essential buses, and external power feeds the left main AC bus. Associated with the external power is a power monitor which measures external power voltage, frequency and phase sequence. Should any one of these parameters be out of tolerance, the monitor prevents closing of the external power contactor. When the external power switch is in the ORIDE position, the external power monitor circuit is bypassed, thus allowing external power which is out of

voltage and frequency tolerance to be applied to aircraft buses. The override position does not override external power with improper phase sequence.

Generator Caution Lamps

Two amber generator caution lamps on the main caution lamp panel illuminate when the respective generator is disconnected from the AC bus. When illuminated, the letters L GEN are visible in the left lamp and R GEN in the right lamp.

ELECTRICAL TEST PANEL

The electrical test panel labeled ELEC TEST, on the aft position of the left console (figure 1-5), consists of a receptacle for plug-in meters and a voltage source selector switch for electrical system maintenance.

CIRCUIT BREAKERS

Individual circuit protection is provided by thermal-type, trip-free circuit breakers. Holding them depressed does not complete the circuit if it remains faulty or overloaded. The main and essential DC buses are protected by 80-amp fuses at the output terminals of T/R's. The crew DC bus is connected to the essential DC bus with a series of 20-amp fuses. The cockpit circuit breaker panel (figure 1-17) is on the aft bulkhead console.

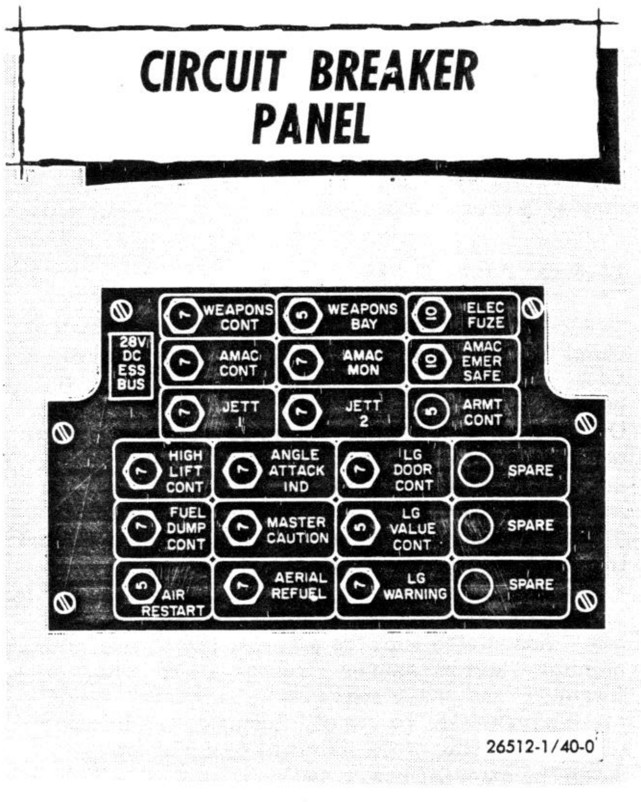

Figure 1-17

ELECTRICAL POWER BUS SOURCE IDENTIFICATION

The power sources for the electrical circuits are from four AC buses and three DC buses. Each circuit is protected by an appropriate circuit breaker. The 115-volt AC buses are identified as left main, right main, essential, and a 26-volt AC instrument bus. The 28-volt DC buses are identified as main, essential, and crew station. In flight, the crew has access only to the crew station circuit breakers.

AC ELECTRICAL POWER BUSES

Left Main

- Aft Utility Outlet
- Auxiliary Flap Power
- Flood Lights, White
- Fuel Pump, Right Outboard (Number 10)
- Fuel Pumps, Numbers 1, 3, and 6
- Forward Utility Outlet
- Inertial Reference Unit Power Number 4
- Inlet Probe Heaters
- Landing/Taxi Light Power
- Left Engine Pressure Ratio Indicator
- Left Engine Nozzle Position Indicator
- Lights, Engine Instruments
- Lights, Flight Instruments
- Lights, Left and Center Console
- Main Transformer-Rectifier (T/R)
- Seat Adjustment (Pilot and MCO)
- Slats/Flaps Emergency Power (Emergency Hi-Lift Power)

Right Main

- Digital Data Communications ASW-27
- Digital Data Indicator
- Formation Lights Power
- Fuel Pump, Left Outboard (Number 9)
- Fuel Pump, Number 2
- HF Communications
- Inertial Reference Unit Power (Number 5)
- Navigation Azimuth Detection
- Navigation System Platform Heaters
- Navigation System Power
- Oil Quantity Indicator
- Pilot Discrete Encoder
- Right Engine Nozzle Indicator
- Right Engine Pressure Ratio Indicator
- Right Main Bus Failure
- Total Temperature Heater
- UHF Data Link ARC-124

Essential

- AC Essential Bus Test
- Aerial Refuel Lamp
- Aircraft Monitor and Control System (AMAC) Power
- Airspeed/Mach Number Indicator
- Air to Ground IFF APX-64
- Anti-Collision Lower and Upper Lights Power
- Approach Power Compensator
- Auxiliary Flight Reference System
- Bay Door (Weapons) Emergency Motor

Central Air Data Computer Power
Data Link Transmitter
Direct View Indicator, VDIG
Electronic Bay Cooling
Essential Transformer-Rectifier Unit (T/R)
External Stores Jettison
Feel and Trim Power
Flaps Position Indicator Flight Control Power A, B, and C
Flood Lights, Red, Left, Right, and Center
Fuel Pumps, Left Inboard (number 7) and Right Inboard (number 8)
Fuel Pumps, Numbers 4 and 5
Fuel Quantity Indicator
Fuel Quantity Selector
Horizontal Situation Indicator (HSI)
Ice Detection
Integrated Armament Control System
Left Engine Fire Detection
Left Engine Fuel Flow Indicator
Left Engine Mach Lever Actuator
Left Generator Test
Lights, Right Console
Left Static Probe Heater
Oxygen Quantity Indicator
Parallel Trim Power
Probe Primary Heater
Probe Secondary Heater
Projection Indicator, VDIG
Radar Altimeter - APN-167
Ram Air Door Actuator
Right Engine Fire Detection
Right Engine Fuel Flow Indicator
Right Engine Mach Lever Actuator
Right Generator Test
Right Pitot/Static Probe Heater
Rudder and Horizontal Stabilizer Position Indicator
Servo Barometric Altimeter
Station 2 through 7 Release Power
Synchro Excitation, Numbers 1 and 2
TACAN
Total Temperature Indicator
UHF/ADF ARA-50
UHF Communications ARC-51B
Wing Selection and Position Indicator
Yaw Series Trim Power
Yaw Trim Control

Twenty Six (26) Volt AC Instrument Bus

Inertial Navigation System HSI
Left Engine Oil Pressure Indicator
Primary Hydraulic Pressure Indicator
Right Engine Oil Pressure Indicator
TACAN
UHF/ADF Antenna
Utility Hydraulic Pressure Indicator

DC ELECTRICAL POWER BUSES

Main Bus

Aft Utility Outlet
Air Curtain Valve
Auxiliary Flap Control
Blanking Pulse J Box
CADC Test
Counting Accelerometer
DC Main Bus Test
Emergency Generator Control
Engine Derichment Control
Engine Exhaust Nozzle Override
Equipment Bay Lights
Forward Utility Outlet
Fuel Pumps, Numbers 9 and 10 (Relays)
Fuel Pumps, Numbers 1, 2, 3, and 6 (Relays)
Ground Ignition Cutoff Switch
Hydraulic Isolation Valve
Inertial Navigation System
Instrument Test
Ladder Switch
Landing/Taxi Light Control
Oil Low Caution Lamp
Radome Control
UHF Data Link Transceiver, ARC-124

Essential Bus

Aerial Refuel Control
Aft Equipment Hot Caution Lamp
Air Conditioning Controls
Air to Ground IFF APX-64
Analog to Analog Converter
Angle-of-Attack-Approach Lamps
Anti Skid Caution Lamp, Brake Valves and Wheel Units
Arresting Hook Control
Arresting Hook Caution Lamp
Attitude Heading and Warning
Automatic Flight Control System Test (AFCS)
Automatic Fuel Control
Augmented Wave Off Control
Automatic Pilot Control
Auxiliary Pitch Trim Control
Auxiliary Receiver ARR-69
Cabin Pressure Caution Lamp
Cabin Pressure Control
Cabin Pressure Warning Lamp
Canopy Warning Lamp
Caution Test (Master Caution Panel, Warning Panel and Fuel Management Panel)
Central Air Datam Computer
Central Air Datam System
DC Essential Bus Test
Direct View Indicator, VDIG
Direct View Indicator, VDIG Servo Unit
Electrical Power Flow Indicator
Engine Anti-Ice
Feel and Trim Power
Fire Detection Test
Flight Control System
Flight Control Test
Fuel Low Caution Lamp
Fuel Manifold Low Fuel Pressure Caution Lamps
Fuel Pumps, Numbers 4, 5, 7, and 8 (Relays)
Fuel Quantity Test
Fuel Tank Pressurization Caution Lamp
Fuel Tank Pressurization Valves
Fuel Wing Outboard Valves
Generator Control
Hydraulic Oil Hot Caution Lamps, Primary and Utility
Hydraulic Oil Shut Off Valves (Fire Pull Handles)

Icing Caution Lamp
Inter Communications (ICS) AIC-25
Landing Gear Position Indicator
Landing Gear Warning Lamp
Landing Gear Valve Control
Launch Bar Control
Launch Bar Warning Lamp
Left Engine Anti-Ice Control
Left Engine Bleed Air Control
Left Engine Fuel Shut Off Valve (Fire Pull Handle)
Left Engine Hydraulic Low Pressure Caution Lamp
Left Engine Oil Cooler Control
Left Engine Oil Hot Caution Lamp
Left Engine Overspeed Caution Lamp
Left Engine Turbine Inlet Temperature Indicator
Left Fire Warning Lamp
Main Landing Gear Safety Relays, Numbers 1 and 2 (Weight on Wheels)
Main Landing Gear Handle Relays, Numbers 1 and 2
Maximum Safe Mach Computer
Mechanical Fuze Stations, Numbers 1 and 8, 2 and 7, and 3 and 6 (Weapons)
Mode Switching Unit
Mode 4, IFF APX-64
Nose Wheel Steering Control
Oil Cooler Ejector Valves
Oxygen Caution Lamp
Projection Indicator
Pitch Damper
Position Lights
Primary Hydraulic Air Shut Off Valve
Probe Heater Control
Probe Heater Caution Lamp
Radar Altimeter APN-167
Right Engine Anti-Ice Control
Right Engine Bleed Air Control
Right Engine Fuel Shut Off Valve (Fire Pull Handle)
Right Engine Hydraulic Low Pressure Caution Lamps
Right Engine Oil Cooler Control
Right Engine Oil Hot Caution Lamp
Right Engine Overspeed Caution Lamp
Right Engine Turbine Inlet Temperature Indicator
Right Fire Warning Lamp
Roll Damper
Servo Barometric Altimeter
Slat/Auxiliary Flap Position Indicator
Speed Brake Control
Speed Brake Position Indicator
Spike Caution Lamps
Spike Emergency Control
Spike Ground Check Buttons
Spoiler Brake Control
Spoiler Monitor
Spoiler Position Indicators
Stations 1 and 8 Power A and B (Weapons)
Stores Refuel Switch
TACAN
Tail Bumper Control
Temperature Control (Crew Module)
Total Temperature Caution Lamp
UHF Communications ARA-50 (D/F)
UHF Communications ARC-51
Utility Hydraulic Air Shut Off Valve
Utility Lights, Pilot and MCO
Weapons Bay Door
Windshield Hot Caution Lamp
Windshield Wash Control
Wing Sweep Indicator
Yaw Damper

Crew Station Bus

Note

The electrical systems connected to the crew station bus are protected by circuit breakers on the aft bulkhead console (figure 1-17). These are the only circuit breakers accessible to the flight crew while in flight. The placarding of each circuit breaker is noted in parentheses following the name of the circuit.

Aerial Refuel Control (AERIAL REFUEL)
Airstart Control (AIR RESTART)
Angle-of-Attack Indicator (ANGLE ATTACK IND)
Aircraft Monitor and Control System Control (AMAC)
Aircraft Monitor and Control System Emergency Safe (AMAC EMER SAFE)
Aircraft Monitor and Control System Monitor (AMAC MON)
Electric Fuze (ELEC FUZE)
Emergency Weapons Bay Door (WEAPONS BAY DOOR)
Fuel Dump Control (FUEL DUMP CONT)
Integrated Armament Control (INT ARMT CONT)
Jettison 1 (JETT 1)
Jettison 2 (JETT 2)
Landing Gear Door Control (LG DOOR CONT)
Landing Gear Valve Control (LG VALVE CONT)
Landing Gear Warning Lamp (LG WARNING)
Master Caution Lamp (MASTER CAUTION)
Slats/Flaps Control (HIGH LIFT CONT)
Weapons Control (WEAPONS CONT)

HYDRAULIC POWER SYSTEM

Hydraulic power is supplied by two independent hydraulic systems, the primary and utility systems (figure 1-18). Both systems operate in parallel to supply hydraulic power for the primary flight controls and wing sweep. In addition to supplying wing sweep and flight control hydraulic power, the utility system also supplies power for operation of the following:

Landing gear	Radome fold
Wheel brakes	Launch bar
Slats and flaps	Emergency electrical generator
Speed brake	Engine air inlet control
Arresting hook	Weapons bay doors
Tail bumber	Flight controls
Nose wheel steering	

If either system should fail, the other is capable of supplying sufficient power for wing sweep and flight control operation. Hydraulic pressure is supplied by four, engine-driven, variable delivery pumps (two for each system). To assure hydraulic pressure if an engine fails, one pump in each system is driven by the right engine, and one pump in each system is driven by the left engine. Either engine can fail and sufficient power will be available for both primary and utility hydraulic systems. Any three pumps can fail and power will still be supplied to the primary flight controls. Pressurized accumulators supplement engine-driven pump delivery during transient hydraulic power requirements. Each system has a piston-type reservoir for hydraulic fluid storage that also acts as a surge damper for return line pressures. These reservoirs are pressurized with nitrogen to insure critical inlet pump pressure for all operating conditions. An automatic isolation valve reserves all utility power output for flight control and wing sweep operation by isolating all other utility functions in the event of primary system failure. Normal isolation shuts off hydraulic power to all utility systems not necessary for flight.

HYDRAULIC PUMPS

Normal power for the primary and utility systems is provided by the engine-driven, variable delivery pumps. One pump in each system is driven by each engine. The pumps are rated at 42.5 gpm. Normal pressure is 3100 psi. Each hydraulic system contains a 15-micron, no-bypass-type filter in each pump discharge line and a 15-micron, bypass-type scavenge line filter.

Hydraulic Handpump

A hydraulic handpump in the main landing gear wheel well replenishes brake accumulator pressure during ground handling operation.

HYDRAULIC ACCUMULATORS

Eight accumulators, three in the primary hydraulic system and five in the utility hydraulic system, are provided. Each system has two accumulators for the horizontal stabilizer actuators and one accumulator for the autopilot damper servos. The utility system has two accumulators for the wheel brake system.

HYDRAULIC FLUID RESERVOIRS

Both primary and utility hydraulic reservoirs are floating piston, air-oil separated type, pressurized at 90 psi. Pneumatic pressure is supplied from pneumatic storage reservoirs, precharged to 500 psi. Engine bleed air is used as an alternate pressure source. A pressure-operated hydraulic relief valve prevents overpressurization by venting excess fluid overboard when reservoir pressure exceeds 135 psi. Steady-state fluid flow is passed through the reservoir to maintain reservoir warmth and to remove air from the fluid. During high flow rates, the fluid is bypassed around the reservoir and cooler loop directly to the pumps by means of a suction bypass valve. A 15-micron, bypass-type filter is located upstream of the reservoir. The reservoir also acts as a surge damper for return line impulse pressure. Capacity of the primary system is three gallons and the utility system is nine gallons.

UTILITY HYDRAULIC SYSTEM ISOLATION VALVE

An isolation valve is incorporated in the utility system to provide automatic and normal isolation of certain functions of the utility system. If there is loss of pressure in the primary system, the valve will automatically reserve all utility power output for flight control and wing sweep operation by isolating all other utility system functions. Normal isolation is selected by the pilot in flight to isolate all utility systems not necessary for flight. This adds a measure of protection if a leak develops in a remote part of the utility system.

Utility Hydraulic System Isolation Switch

The utility hydraulic system isolation switch with positions marked ON and ISOL is on the landing gear panel (figure 1-25). The ON position supplies hydraulic pressure to all utility hydraulic system components, provided there is no drop of primary system pressure below approximately 400 psig. When in the ISOL position, the following systems are isolated:

Landing gear	Wheel brakes
Arresting hook	Radome fold
Tail bumper	Launch bar
Nose wheel steering	

When the landing gear handle is moved to DN, the isolation switch is automatically moved to the ON position.

HYDRAULIC COOLING

Cooling is provided by an air-to-hydraulic heat exchanger and a fuel-to-hydraulic heat exchanger in each hydraulic system. Hydraulic fluid temperature is limited to 275° F (135°C) at pump inlets. The

cooling medium is air only at low speeds, fuel and air at intermediate speeds, and fuel only at high speeds. Cooling air flow on the ground is provided by an ejector device powered by engine bleed air.

Hydraulic Fluid Overheat Caution Lamps

Two hydraulic fluid overheat caution lamps, one for each system, are on the main caution lamp panel (figure 1-28). The lamps illuminate when the hydraulic fluid temperature of the associated system exceeds 230 ± 10°F (110 ± 6°C). When illuminated, the respective lamps display PRI HOT, and UTIL HOT.

HYDRAULIC PRESSURE INDICATORS

There are two 0 to 4000 psi hydraulic pressure indicators on the forward end of the left console, one each for the primary and the utility system. Pressure is measured mechanically and transmitted electrically by pressure transmitters in the system pressure lines.

Hydraulic Low-Pressure Caution Lamps

Four low-pressure caution lamps, one for each pump are on the main caution lamp panel (figure 1-28). These lamps illuminate when individual pump discharge pressure falls below 500 ± 100 psi. When illuminated, the respective lamps display L PRI HYD, L UTIL HYD, R PRI HYD, and R UTIL HYD.

PNEUMATIC POWER SUPPLY SYSTEMS

There are three independent pneumatic power supply systems which provide pressure for emergency operation of the landing gear, emergency operation of the engine inlet control system (inlet spikes), and normal pressurization of the hydraulic reservoirs. There is one 450-cubic-inch bottle for the landing gear and two parallel 250-cubic-inch bottles for the inlet spikes. These three bottles are pressurized to 3000 psi through check valves from one external connection. Each system has a pressure gage. Charge gas may be air; however, nitrogen is preferred. Pressurization of the primary and utility hydraulic system reservoirs is normally provided by a gas storage container in each reservoir. Engine bleed air is available through a shuttle valve if stored air pressure falls below the pressure of the engine bleed air. Bleed air passes through a moisture trap and chemical dryer before entering the shuttle valve.

FLIGHT CONTROL SYSTEM

The flight control system (figure 1-19) provides control of the aircraft through movement of the horizontal stabilizers, spoilers, and rudder, using conventional stick and rudder pedal cockpit controls. Rate gyros and accelerometers, in conjunction with electronic computers, provide continuous automatic damping about the three axes of the aircraft. Three separate channels, pitch, roll, and yaw direct hydraulic servo actuators that control surface movement. Pitch attitude of the aircraft is controlled by symmetrical deflection of the horizontal stabilizer surfaces. Roll attitude is controlled by asymmetrical deflection of the horizontal stabilizer surfaces, which are augmented by two spoiler segments on top of each wing when wing sweep angle is less than 45 degrees. Aircraft yaw control is accomplished by deflection of a rudder surface on the trailing edge of the vertical stabilizer. Stability augmentation is provided for pitch, roll, and yaw by triple-redundant sensors, electronic circuitry, and electro-hydraulic damper servos. Automatic failure detection and self-test features are also provided. The control stick and rudder pedals are mechanically connected to hydraulic servo actuators at empennage control surfaces. The pitch-roll mixer translates pitch and roll commands into left and right horizontal stabilizer commands to the servo actuators. The servo actuators are supplied pressure from both primary and utility hydraulic systems, but are fully operable with the loss of one hydraulic system.

PITCH

Manual control of the aircraft in pitch is achieved by fore and aft movement of the control stick. This movement is transmitted along pitch channel push-pull tubes and bellcranks, through the pitch-roll mixer, to left and right horizontal stabilizer actuator control valves. These control valves control hydraulic pressure to the actuators, thus causing horizontal stabilizer control surfaces to move symmetrically. Stick throw is limited by mechanical stops.

Pitch Trim

Pitch trim can be controlled by either the trim button on top of the control stick grip (figure 1-20) or the auxiliary pitch trim switch on auxiliary flight control panel (figure 1-21). The trim button can control trim in either a parallel or a series mode. The auxiliary pitch trim switch can only control trim in the series mode.

Pitch Parallel Trim

Pitch parallel trim is controlled by the trim button on top of the control stick grip when the pitch damper switch on autopilot damper panel (figure 1-22) is in DAMPER and auxiliary pitch trim switch is in STICK position. Parallel trim changes will move control stick, therefore trim rates and authority will vary through the command augmentation as a function of pitch gain.

Pitch Series Trim

Pitch series trim is controlled by the trim button on top of the control stick grip when the pitch damper switch is in OFF position and auxiliary pitch trim switch is in STICK. If auxiliary pitch trim switch is not in the STICK position and pitch damper switch is

HYDRAULIC POWER (UTILITY)

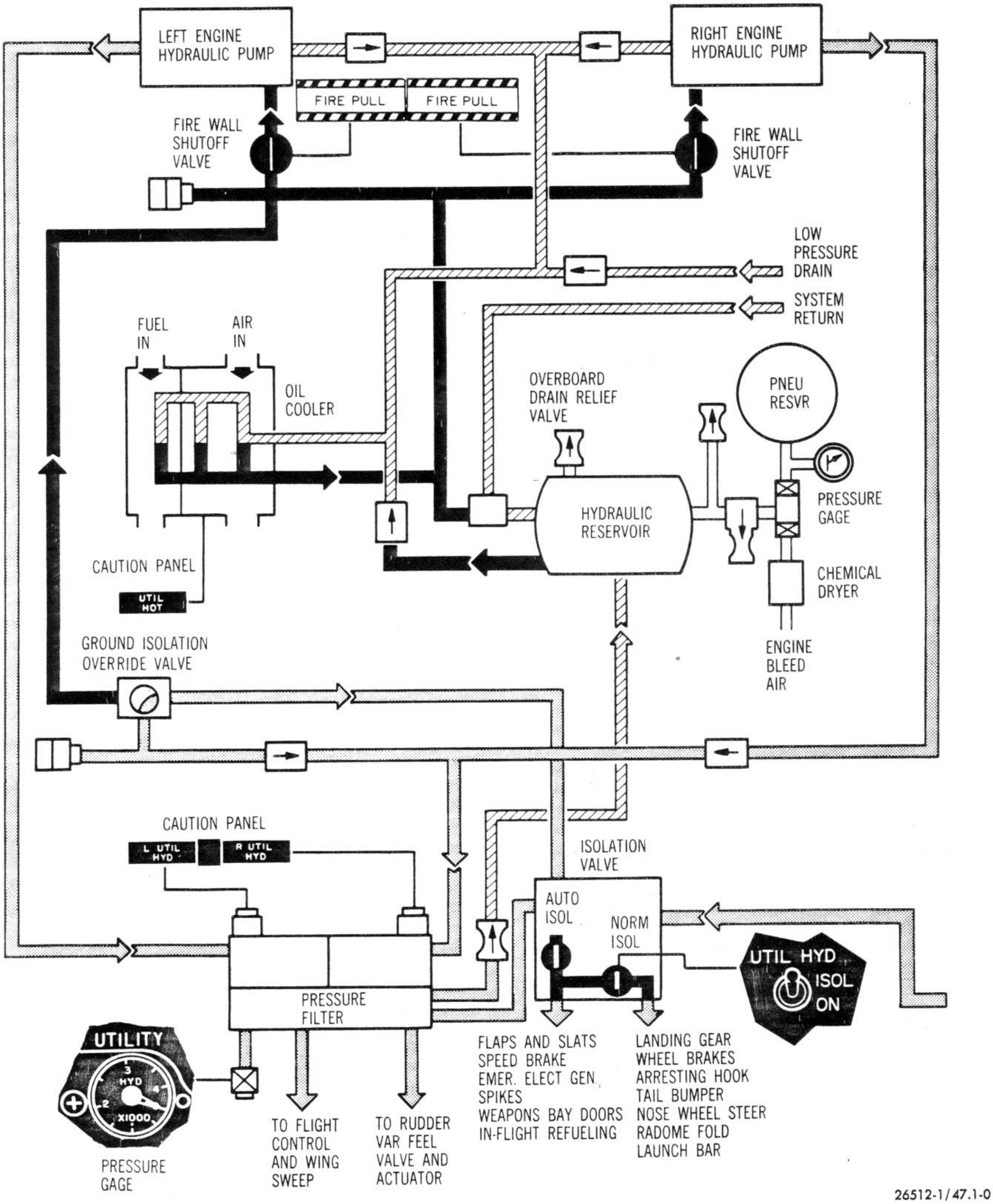

Figure 1-18 (Sheet 1)

HYDRAULIC POWER (PRIMARY)

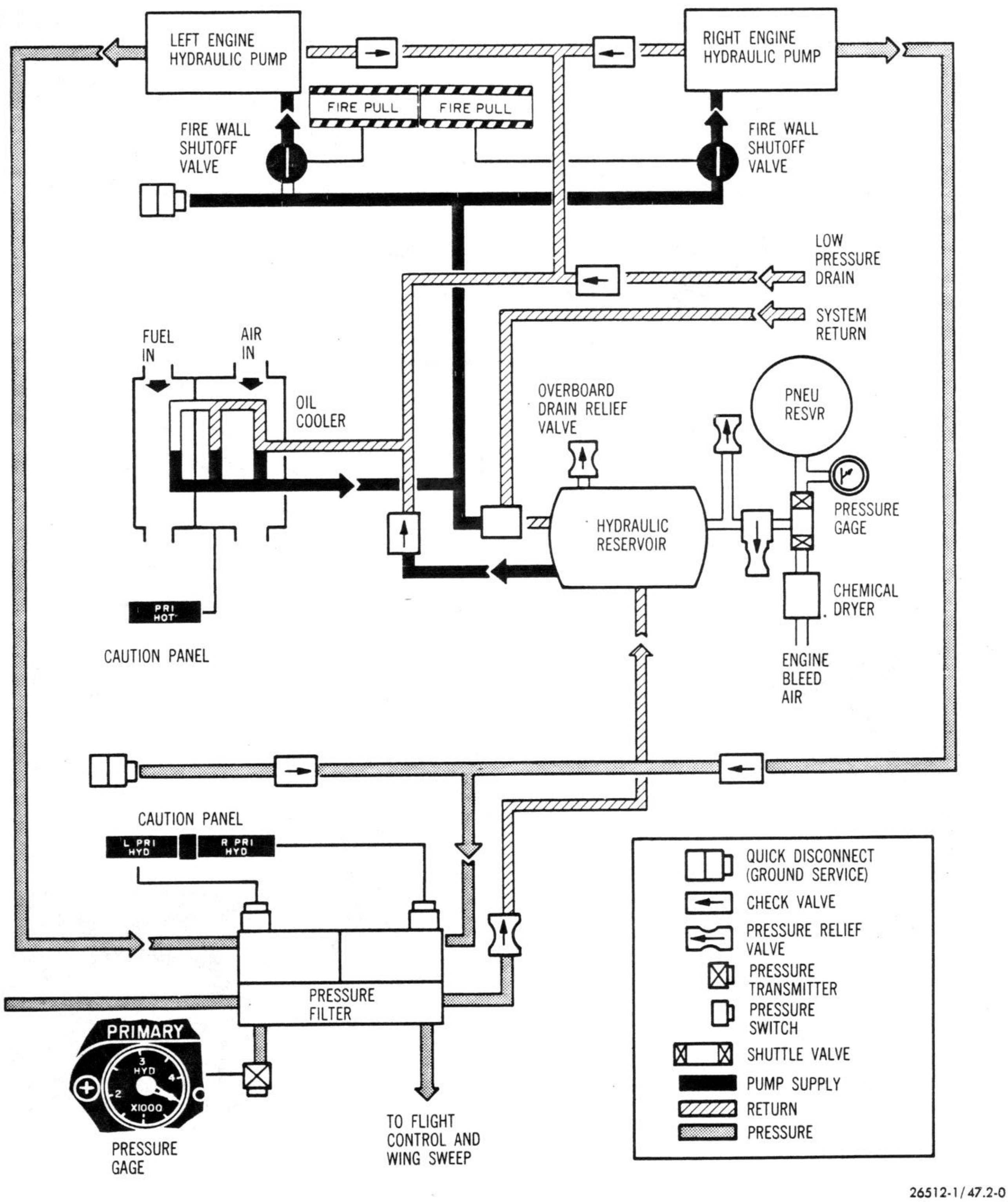

Figure 1-18 (Sheet 2)

FLIGHT CONTROL SYSTEM

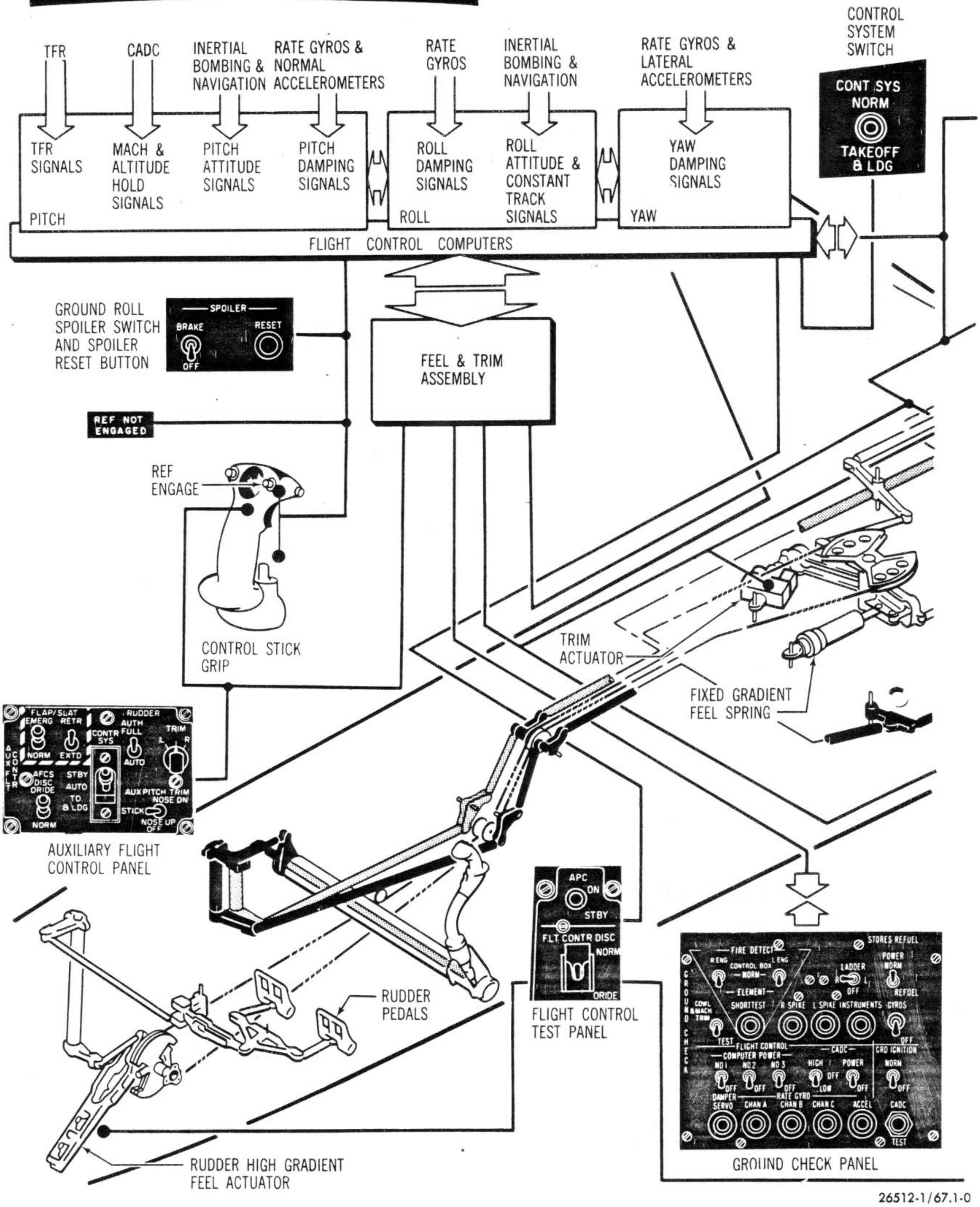

Figure 1-19 (Sheet 1)

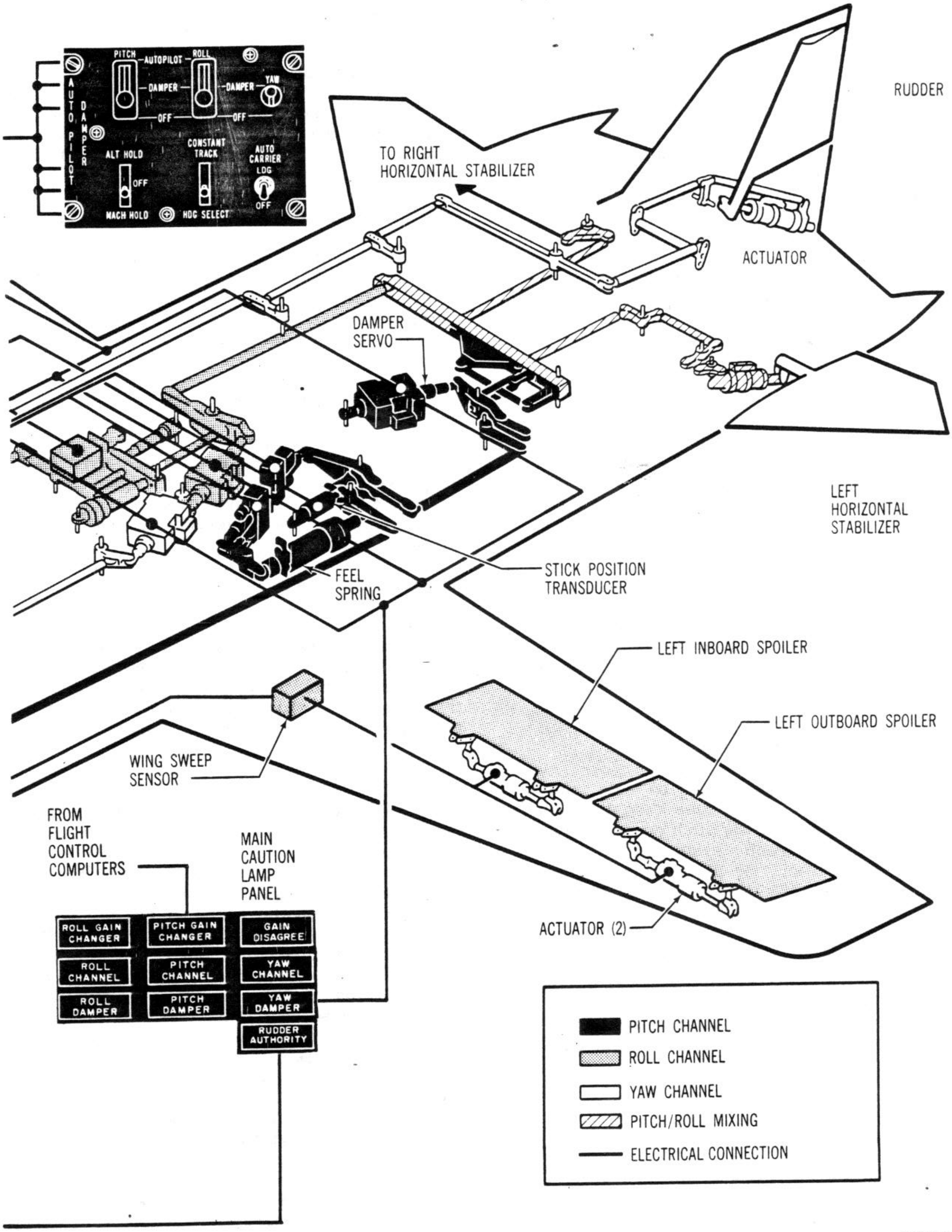

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Figure 1-19 (Sheet 2)

OFF. series trim is deactivated as well as parallel trim from the trim button on the stick grip; but series trim is available and can be controlled with auxiliary pitch trim switch. Changes in trim through the series mode will not move control stick.

Pitch Series Trim Followup

Pitch series trim followup is incorporated into system to ensure full pitch damper authority during sustained maneuvers with pitch stability augmentation. Displacement of control stick causes a signal to displace and aid mechanical stick command. The damper displacement is sensed and fed to pitch series trim actuator through series trim computer in feel and trim assembly. The actuator moves elevator linkage an amount proportional to and in same direction as displacement of damper, thus allowing damper to return to its neutral position. Pitch series trim followup functions only when slats are up and flight control switch is in AUTO.

Pitch Channel Caution Lamp

The pitch channel caution lamp on main caution lamp panel (figure 1-28) illuminates when a malfunction is sensed in the pitch channel computer. Since the electronics system is triple-redundant, illumination of this lamp does not necessarily indicate a complete failure.

Pitch Damper Caution Lamp

The pitch damper caution lamp on main caution lamp panel (figure 1-28) illuminates when a malfunction is sensed in the pitch damper. Since damper has two active valves and a model valve, the illumination of this lamp does not necessarily indicate complete damper failure.

Pitch Gain Changer Caution Lamp

The pitch gain changer caution lamp on main caution lamp panel (figure 1-28) illuminates when a malfunction has been sensed in the pitch computer gain circuit. Since the electronics system in this computer is triple-redundant, the illumination of this lamp does not necessarily indicate a complete failure.

Stall Warning System

The stall warning system shakes the rudder pedals to provide additional warning that the aircraft is approaching a stall. The system is automatically activated when aircraft exceeds 19 (±1) degrees angle-of-attack with slats extended.

ROLL

Manual control of the aircraft in roll is achieved by lateral movement of the control stick. These movements are transmitted along the roll channel push-pull tubes and bellcranks, through the pitch-roll mixer to the horizontal stabilizer actuator control valves. These valves operate the horizontal stabilizer actuators in opposite directions, causing an asymmetrical movement of the horizontal stabilizer control surface.

Roll Trim

Roll trim is accomplished through the roll damper servo and therefore, roll trim is lost when roll damper is off. Since the output of the roll damper servo is in series with the roll channel linkage, the control stick does not move as trim is applied. Roll trim is controlled by a trim button on the top of the control stick.

Roll Channel Caution Lamp

The roll channel caution lamp on main caution lamp panel (figure 1-28) illuminates when a malfunction is sensed in the roll channel computer. Since the electronics system is triple-redundant, illumination of this lamp does not necessarily indicate a complete failure.

Roll Damper Caution Lamp

The roll damper caution lamp on the main caution lamp panel (figure 1-28) illuminates when a malfunction is sensed in the roll damper. Since damper has two active valves and a model valve, the illumination of this lamp does not necessarily indicate a complete damper failure.

Roll Gain Changer Caution Lamp

The roll gain changer caution lamp on main caution lamp panel (figure 1-28) illuminates when a malfunction has been sensed in the roll computer circuit. Since the electronics system in this computer is triple-redundant, illumination of this lamp does not necessarily indicate a complete failure.

Spoilers

When wings are forward of 45 degrees, roll control is aided by action of two spoilers on top of each wing. Each spoiler surface is actuated by a hydraulic servo actuator. The outboard spoiler actuators are supplied driving pressure by the utility hydraulic system, and the inboard spoiler actuators are supplied driving pressure by the primary hydraulic system. If either hydraulic system fails, the other system will drop and lock the affected spoiler segments in the down position. The actuators receive their command signals from transducers in the roll channel linkage. Lateral movement of the control stick causes transducers in the stick position transducer assembly to generate common signals which are sent through wing sweep sensor assembly to spoiler actuators. The spoilers extend to a maximum of 45 degrees in response to one half lateral stick displacement (force detent). The spoiler command to stick position is nonlinear (low gradient through neutral). The spoilers are operated only when wing sweep angle is between 16 and 45 degrees. When the wing sweep angle is 45 degrees, the inboard spoiler command signals are zeroed and spoilers are locked down. At 47 degrees, outboard spoilers are locked down in

same manner, and hydraulic pressure is shut off to inboard spoilers. At 49 degrees, the hydraulic pressure is shut off to outboard spoilers. On the ground, the flight control spoilers can be symmetrically raised for aerodynamic braking during landing roll. In flight, if either an inboard or an outboard pair of spoilers simultaneously extend 15 degrees or more, a failure detection network will drop and lock both segments in down position. The SPOILER caution lamp on main caution panel will illuminate to indicate this failure. The other pair of spoilers will remain operational. The failed pair of spoilers and caution lamp may be reset by depressing SPOILER RESET button on left sidewall.

Spoiler Caution Lamp

The spoiler caution lamp on main caution lamp panel (figure 1-28) will illuminate when a malfunction in spoiler control circuitry occurs, causing a symmetric pair of spoilers to be locked down.

Spoiler Reset Button

The spoiler reset button on left sidewall (figure 1-5) is a momentary pushbutton, placarded SPOILER RESET. The button is for resetting spoiler circuitry if a malfunction has caused a pair of spoilers to be locked down. If a pair of spoilers has been locked down and the spoiler caution lamp is illuminated, depressing the spoiler reset button will cause spoiler caution lamp to go out and spoiler circuitry to be reset, enabling it to accept signals from spoiler transducers.

Spoiler Position Indicator

The position of spoilers is indicated on four flip-flop type indicators, two for left segments and two for right segments. The indicators are on the control surface position indicator on left console (figure 1-5) under landing gear panel. When a segment is down, letters DN appear in individual window. As segments extend, the individual window becomes blank.

Spoiler Self-Test Switch

The spoiler self-test switch on left console (figure 1-5) is placarded SPOILER TEST and marked INBD, OFF and OUTBD. Selecting outboard will cause outboard spoilers to raise, then lower and lock down. Lateral control stick inputs will not cause the outboard spoilers to deflect in this induced failed mode until spoiler reset button has been depressed. However, should switch be moved to select inboard, causing inboard spoilers to similarly fail, outboard spoilers will become operative. The spoiler caution lamp will illuminate during this test.

PITCH-ROLL MIXING

The combined roll and pitch movements of the control stick are transmitted by the linkage of their respective channels to the pitch-roll mixer assembly, where they are combined and converted into left and right horizontal stabilizer actuator command signals. The mixer pitch channel input stops are set at 25 degrees up and 15 degrees down symmetrical horizontal stabilizer command. The mixer roll channel input stops are set at 8 degrees of lateral command. The combined mixer stops limit individual actuator commands to 31-1/2 degrees up or 15-1/2 degrees down.

YAW

Manual control of the aircraft in yaw is achieved by using conventional rudder and rudder pedals. Movement of rudder pedals is transmitted to the rudder actuator control valve by a combination of control cables, push-pull tubes, and bellcranks. The control valve controls the flow of the hydraulic fluid to rudder actuator. The actuator moves rudder in the direction commanded by rudder pedals. When the aircraft is in the landing configuration, rudder pedal travel and rudder movement is unrestricted. In-flight configuration restricts rudder movement. To aid in spin recovery, unrestricted rudder movement is available.

Yaw Trim

Yaw trim is accomplished by an electrically driven actuator which mechanically positions rudder linkage. Since yaw trim actuator is in series with rudder linkage, there is no movement of rudder pedals as trim is applied. Yaw trim is controlled by a rudder trim switch on the auxiliary flight control panel (figure 1-21).

Rudder Pedals

Conventional rudder pedals are used for yaw control by controlling rudder and for taxiing by controlling nose wheel steering. The pedals also accomodate conventional toe brake pedals that are mechanically connected to brake metering valves.

Yaw Channel Caution Lamp

The yaw channel caution lamp on main caution lamp panel (figure 1-28) illuminates when a malfunction is sensed in the yaw channel computer. Since the electronics system is triple-redundant, illumination of this lamp does not necessarily indicate a complete failure.

Yaw Damper Caution Lamp

The yaw damper caution lamp on the main caution lamp panel (figure 1-28) illuminates when a malfunction is sensed in the yaw damper. Since damper has two active valves and a model valve, the illumination of this lamp does not necessarily indicate a complete failure.

ADVERSE YAW COMPENSATION

An adverse yaw compensation system is incorporated in the flight control system to enhance coordination in turns when the aircraft is in the landing configuration. The system is activated when the slats are extended. When the system is activated, side slip angle, washed out yaw rate and roll rate signals are

sent to the yaw damper. The roll rate signal gain increases in proportion to angle-of-attack and moves the rudder in the direction of the roll command. The side slip angle signal moves the rudder in the direction required to return the side slip angle to zero. The system can be turned off by placing the AFCS DISC switch to ORIDE.

LOW SPEED TRIM COMPENSATION

Low speed trim compensation is incorporated in the flight control system to provide increased low speed stability when the aircraft is in the landing configuration. The system is referenced at 9 degrees angle-of-attack and is activated when the slats are extended. As angle-of-attack is increased above 9 degrees, a signal is automatically sent to displace the pitch damper down. For a decrease in angle-of-attack the damper movement is up. Damper movement is one degree for each degree of angle-of-attack. The signal is faded into the damper gradually to reduce engage or disengage transients. The pilot must offset this movement with stick command in order to hold one g flight. The result is a more apparent change in stick force with speed variation.

CONTROL STICK

The control stick is mechanically connected to the hydraulic servo actuators at the empennage control surfaces, thereby providing pilot with manual pitch and roll control. The stick grip (figure 1-20) contains a trim button, weapon release button, reference engage button, nose wheel steering button, and an autopilot release lever. It can also be used in emergencies as a means of actuating crew module bilge/flotation bag inflation pump.

Pitch, Roll Trim Button

A trim button on the control stick grip (figure 1-20) is for normal manual trim of pitch and roll axes. The button has five positions—up, down, left and right. and is spring-loaded to the center off position. Moving button aft (NOSE UP) or forward (NOSE DOWN) causes the trailing edges of the horizontal stabilizer surfaces to move symmetrically up or down, respectively. When pitch damper is in DAMPER position, trim button operates the parallel pitch trim mode and control stick will move as the trim is changed. When pitch damper is in OFF position, trim button operates series trim mode and control stick does not move as trim is changed. Moving trim button left (LWD) or right (RWD) causes trailing edges of horizontal stabilizer surfaces to move asymmetrically up and down as selected. Roll trim is a series function and does not move the control stick.

Weapon Release Button

This switch has no operational function.

Reference Engage Button

For information, refer to Autopilot, this section.

Nose Wheel Steering Button

For information, refer to Landing Gear, this section.

Autopilot Release Lever

For information, refer to Autopilot, this section.

NULL TRIM BUTTON

The null trim button (figure 1-5) on left console is for setting the control surfaces to a take-off configuration. When it is depressed, the following occurs: null trim relay is energized; pitch parallel trim and yaw trim actuators are driven to zero degrees; roll integrator is synchronized so that output to roll damper is zero; pitch trim integrator is driven to a null; and pitch trim series actuator is driven to a nose-up positon of 3.8 degrees.

Null Trim Lamp

The null trim lamp on landing gear panel (figure 1-25) on left console illuminates when all trim actuators are at their null position.

CONTROL SURFACE POSITION INDICATOR

The control surface position indicator on left console (figure 1-5) under landing gear panel is composed of three sets of indicators. These indicators show position of spoilers, rudder, and horizontal tail (horizontal stabilizer). The position of spoilers is

Figure 1-20

indicated on four flip-flop type indicators—two for the left and two for the right spoilers. When spoilers are retracted, letters DN appear in each window. As spoilers extend, the window becomes blank. Rudder position is shown by a pointer on a scale, graduated in 5 degree increments, 30 degrees left or right of zero. The position of horizontal stabilizers is indicated by two pointers marked L and R, on a scale 30 degrees up and 20 degrees down. The scale is graduated in 2 degree increments. Another pointer indicates left wing down or right wing down.

FLIGHT CONTROL MASTER TEST BUTTON

The flight control master test button on left console (figure 1-5) provides a source of power to flight control test switches and buttons on ground check panel. Depressing button closes a switch, thus allowing power to be applied to flight control test switches and buttons. When button is released, these switches and buttons are inoperable. This switch is interlocked with weight on wheels switch to prevent operation of test functions in flight.

STABILITY AUGMENTATION TEST SWITCH

The stability augmentation test switch on left console (figure 1-5) is a three-positioned switch marked SURF MOTION AND LIGHTS, SURF MOTION, and OFF. The switch provides a means of ground checking the stability augmentation system. With the flight control switch in TO & LD, placing the switch to SURF MOTION causes the horizontal stabilizers to move to a left roll, nose down position, and rudder to move to right. Placing the switch to SURF MOTION AND LIGHTS causes the horizontal stabilizers to move to a left roll, nose down position; rudder to move to right, three damper and three channel and two gain changer lamps to illuminate. With the flight control switch in AUTO, the tests will be the same for pitch and roll but rudder will move right then left for the SURF MOTION test, and right then center for the SURF MOTION AND LIGHTS test because of the yaw rate washout effects. For either test position, the lamps may not illuminate unless the gain is at 100 percent. The flight control master test button must be depressed to obtain these test functions.

AUXILIARY FLIGHT CONTROL PANEL

The auxiliary flight control panel (figure 1-21) placarded AUX FLT CONTR on left console provides an auxiliary pitch trim switch, rudder authority trim switches, a flight control system switch, and an automatic flight control system switch. The emergency slats/flaps extension switches and an emergency autopilot override switch (AFCS) are also on this panel and are discussed, respectively, under Slats/Flaps and Autopilot, this section.

Auxiliary Pitch Trim Switch

An auxiliary pitch trim switch placarded STICK, NOSE DN, NOSE UP, and OFF is on auxiliary flight control panel (figure 1-21). The switch controls pitch trim series actuator, and pitch damper. It is

AUXILIARY FLIGHT CONTROL PANEL

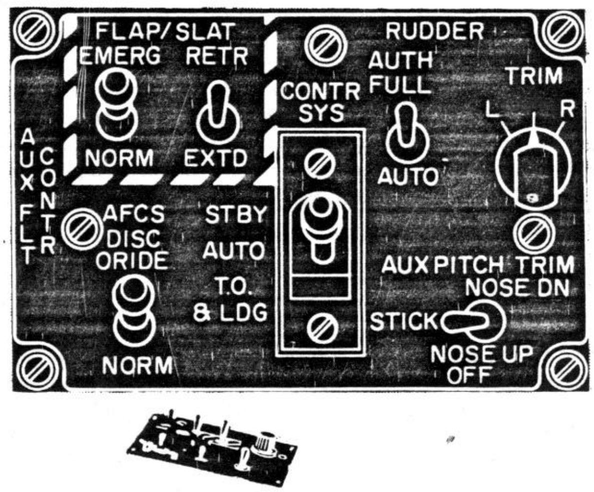

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Figure 1-21

a four-positioned switch, spring-loaded to center OFF from NOSE DN and NOSE UP. In STICK, pitch trim signals can be commanded only by the trim button on control stick. When the switch is held NOSE DN or NOSE UP, pitch series trim actuator and pitch damper move the horizontal stabilizer symmetrically, as selected, until switch is released to OFF or limits are reached. With the switch in OFF, NOSE DN or NOSE UP, trim button on control stick is inoperative.

Note

The auxiliary pitch trim switch should be left in STICK at all times for normal operation. Since the auxiliary pitch trim serves as a backup trim system, it should only be used when normal trim operation fails.

Rudder Trim Switch

The rudder trim switch on auxiliary flight control panel (figure 1-21) is for rudder trim control. The switch has two positions placarded L and R and spring-loaded to the center which is an unmarked off position. Rotating switch to L or R causes rudder trim actuator to drive the rudder to selected direction until switch is released or a maximum of 10-1/4 degrees rudder deflection is reached.

Rudder Authority Switch

The rudder authority switch on auxiliary flight control panel (figure 1-21) has two positions placarded FULL and AUTO. In AUTO, full rudder authority of 30 degrees either side of center is available when flight control switch is in AUTO and the landing gear handle and the slats/flaps handle are down. Full rudder authority is also available any time the flight control switch is positioned to TO & LDG. With the flight control switch still in AUTO, rudder authority changes to 11 degrees either side of center when the landing gear handle and the slats/flaps handle are in the up position. When rudder authority switch is in FULL, full rudder authority is available regardless of flight control switching. This is provided to assist spin recovery.

Flight Control Switch

The flight control switch on auxiliary flight control panel (figure 1-21) is placarded CONT SYS and has three positions placarded TO & LDG, AUTO, and STBY. The switch should be placed in AUTO for all normal operations. When the landing gear and the slats/flaps are lowered, a take-off and landing configuration is achieved and the following actions occur: the pitch and roll computer gains are set at 30 percent and 100 percent, respectively; rudder authority becomes 30 degrees; series trim is locked, normal acceleration feedback is locked out; yaw rate gain and washout time constant are selected; and power is applied to low speed trim compensation and adverse yaw compensation circuits. Raising the landing gear and slats/flaps selects in-flight configuration again. The TO & LDG is an emergency position which selects the flight control system to a take-off and landing configuration. This position is used when GAIN DISAGREE caution lamp illuminates on main caution panel. The STBY selects 10 percent and 20 percent pitch and roll computer gains, respectively, and is used when operating on a single hydraulic system.

AUTOPILOT SYSTEM

The autopilot system consists of electronic circuitry that, in conjunction with primary flight control system, controls aircraft during five modes of autopilot flight. The autopilot system receives input signals from other systems and computes command signals to pitch and roll dampers to control aircraft. The autopilot modes are: attitude stabilization; mach hold; altitude hold; constant track and heading select. Incompatible mode selection is prevented by circuit interlocks. Attitude stabilization is normally in effect when autopilot is engaged. Attitude stabilization will hold aircraft at the reference roll or pitch attitude or both until selection of another autopilot mode, or until pilot initiation of control stick steering. The aircraft may be manually maneuvered at any time by use of control stick steering without disengaging the autopilot. During operation of the autopilot, control stick will not follow movement of the surfaces. Pitch autopilot disengage transients are reduced by pitch series trim actuator. Engage transients are prevented by continuous synchronization of pitch and roll attitude input signals so that their commands to dampers are zero at time of autopilot engagement.

ATTITUDE STABILIZATION MODE

The attitude stabilization mode is the initial control mode established when the autopilot is engaged. Attitude stabilization can be engaged in either roll or pitch channel or both. Attitude reference signals are received by pitch and roll computers from the inertial reference unit of the inertial navigation system (INS). Resultant signals from the pitch and roll computers control the pitch and roll dampers, thus holding the aircraft at reference attitude existing at time of autopilot engagement; however, roll angles of less than 3 (±1) degrees will result in a wings level attitude command upon engagement of this mode. If constant track mode is selected, roll damper will control aircraft according to new reference. However, when constant track mode is discontinued, autopilot will revert back to attitude stabilization and will maintain attitude that existed at time of disengagement. For example, if pitch and roll autopilot are engaged with aircraft in a 20 degree bank, this bank angle will be held. If constant track mode is then selected, aircraft will respond by returning to selected ground track. The original pitch attitude will continue to be controlled by attitude stabilization and will remain unchanged. If constant track mode is subsequently discontinued, autopilot will revert back to attitude stabilization in roll. The pilot may change attitude stabilization pitch and roll references at any time by using control stick steering. The autopilot emergency override (AFCS) switch on the auxiliary flight control panel (figure 1-21) must be in NORM to engage autopilot.

MACH HOLD MODE

The mach hold mode maintains constant mach. In this mode, throttle position is fixed and mach is controlled by aircraft pitch attitude through operation of horizontal stabilizer surfaces. Upon engagement of this mode, a mach reference is set up in the central air data computer (CADC). Any deviation of mach from this reference results in an error signal being set to the pitch computer from CADC. If mach increases above reference, resulting mach error signal will command a nose-up attitude through pitch damper, causing aircraft to return to referenced mach number. An opposite command is used for a decrease in mach.

ALTITUDE HOLD MODE

The altitude hold mode automatically maintains constant altitude. Upon engagement of this mode, an altitude reference is established in central air data computer (CADC). Any deviation in altitude by aircraft results in an altitude error being fed to the pitch computer from CADC. If aircraft altitude increases above reference, resulting altitude error signal will command a nose-down altitude through pitch damper until desired altitude is obtained. An opposite command is given for a decrease in altitude.

CONSTANT TRACK MODE

The constant track mode maintains aircraft on a constant ground track. When this mode is engaged, existing ground track is sensed in the inertial navigation system and is set up as a mode reference. Any deviation from this reference by aircraft results in an error signal being sent from the inertial navigation system to roll computer. The roll computer, in turn, sends a command to roll damper, correcting the deviation.

HEADING SELECT NAVIGATION MODE

When operating in the heading select navigation mode, roll computer receives steering error signals from the inertial navigation system (INS) to steer the aircraft to a preset destination set into the INS destination counters. The steering error signal is obtained by comparing actual ground track with the computed course—from present position to destination. The computed course may be either great circle or short range depending on position of INS mode selector switch. The roll computer, in turn, sends commands to the roll damper to turn aircraft to fly to destination.

CONTROL STICK STEERING

When any autopilot mode is engaged, including basic attitude stabilization, attitude reference controlling the aircraft can be disengaged by use of control stick steering. Control stick steering is activated in the pitch channel by applying a force greater than 1.7 pounds, in a forward or aft direction, to top of control stick. This mode is activated in the roll channel by applying a force of 1.3 pounds laterally to control stick. When this force is applied to either or both channels, reference or references are disengaged, a caution lamp will illuminate, and pilot can maneuver aircraft to a new reference. When force to the control stick is reduced below 1.7 pounds in the pitch channel or 1.3 pounds in the roll channel, attitude stabilization will automatically reengage in the affected channel or channels, provided attitude limits are not exceeded. The reference engage button must be depressed to reengage autopilot. The attitude limits are ±30 degrees in pitch and ±60 degrees in roll. Should these limits be exceeded in one or both channels, attitude stabilization will not reengage in that channel until its attitude angle is reduced to less than its limit. In addition, roll channel can not be engaged if either pitch attitude is greater than ±30 degrees or the yaw damper is turned OFF.

AUTOPILOT DAMPER PANEL

The autopilot damper panel (figure 1-22) on center console provides switches for selecting autopilot modes of operation.

Autopilot/Damper Switches

Three damper switches, one each for pitch, roll, and yaw channels, are on the autopilot damper panel (figure 1-22) on the center console. The pitch and roll damper switches are three-positioned switches

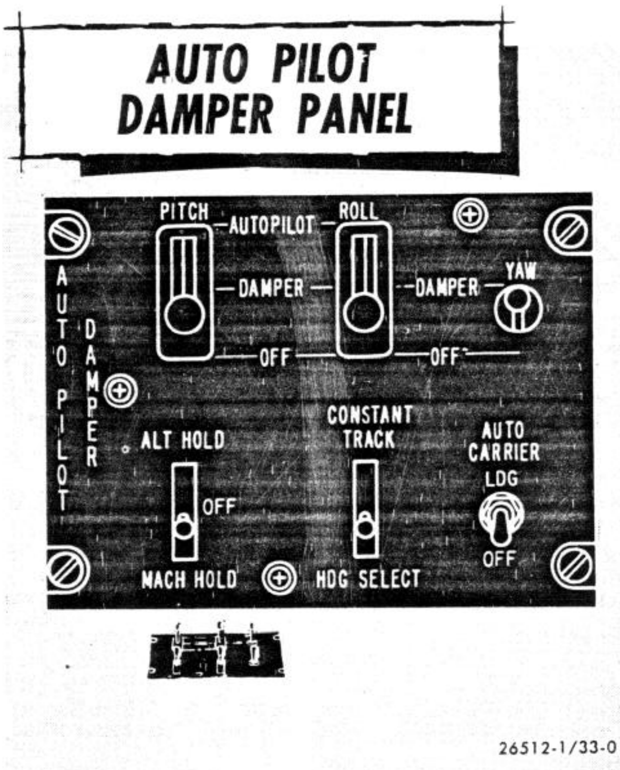

Figure 1-22

placarded AUTOPILOT, DAMPER, and OFF. They are solenoid-held in AUTOPILOT and OFF and are spring-loaded to the damper position. The yaw damper is a two-positioned switch placarded DAMPER and OFF. It is also solenoid-held in OFF and spring-loaded to DAMPER. Placing any of the switches to DAMPER engages damper and provides stability augmentation for the respective channel. Placing a switch to OFF disengages damper of the respective channel and causes the respective damper caution lamp on main caution panel to illuminate. Placing either pitch or roll switch to AUTOPILOT will engage autopilot attitude stabilization.

Constant Track/Heading Select Switch

The constant track/heading select switch on autopilot damper panel (figure 1-22) is a three-positioned switch marked CONSTANT TRACK, HDG SELECT, and an unmarked center off position. The switch is solenoid-held by 28-volt DC power to CONSTANT TRACK or HDG SELECT and is spring-loaded to off. When the switch is placed in CONSTANT TRACK and reference engage button is depressed, aircraft will be held on a constant ground track. When the switch is placed in HDG SELECT and the reference engage button is depressed, aircraft will be held on course to the destination set in the navigation system. The switch will not latch in CONSTANT TRACK or HDG SELECT unless the roll autopilot/damper switch is

in AUTOPILOT. If while operating in CONSTANT TRACK or HDG SELECT, 28-volt DC power to the holding solenoid is lost, the switch will return to the center off position. The REF NOT ENGAGED lamp will illuminate, indicating this malfunction. When the switch is positioned to the center off position, autopilot will discontinue controlling aircraft and it will revert to attitude stabilization in the roll channel.

Altitude Hold/Mach Hold Selector Switch

The altitude hold/mach hold selector switch (figure 1-22) on the autopilot damper panel is a three-positioned switch marked ALT HOLD, OFF, and MACH HOLD. The switch is solenoid-held by 28-volt DC power to ALT HOLD or MACH HOLD and is spring-loaded to OFF. When the switch is in the ALT HOLD position and the reference engage button is depressed, the autopilot will control aircraft to maintain the altitude present at time mode was engaged. When the switch is positioned to MACH HOLD and the reference engage button is depressed, the autopilot will control aircraft to maintain the mach number present at time of mode engagement. The switch will not latch in either ALT HOLD or MACH HOLD position if pitch autopilot/ damper switch is not in AUTOPILOT position. If while operating in MACH HOLD or ALT HOLD position, 28-volt DC power to hold relay is lost, switch will return to OFF. The reference not engaged lamp will not illuminate for this malfunction.

Automatic Carrier Landing Switch

This switch has no operational function at this time.

REFERENCE ENGAGE BUTTON

A reference engage button marked REF ENGAGE is on control stick grip (figure 1-20). When any autopilot mode is selected, other than attitude stabilization, reference engage button must be depressed before mode will engage.

Reference Not Engaged Caution Lamp

The reference not engaged caution lamp (figure 1-28) on pilot's instrument panel illuminates under the following conditions: when the autopilot/damper switches are in the autopilot position and control stick steering is being used, when any autopilot mode (altitude hold, mach hold, constant track or heading select) is selected and the reference engage button has not been depressed; and when the emergency autopilot switch (AFCS) is placed in ORIDE. The letters REF NOT ENGAGED will be visible in face of lamp when illuminated.

Note

> The use of control stick steering after any autopilot mode has been engaged will result in mode being disengaged, and lamp will illuminate and remain on until reference engage button is depressed again.

AUTOPILOT RELEASE LEVER

The autopilot release lever on base of stick grip (figure 1-20) permits pilot to disengage autopilot without removing his hand from control stick. Depressing lever will return autopilot/damper switches to DAMPER. This disengages all autopilot functions and places aircraft under pilot control.

AUTOPILOT FLIGHT CONTROL (AFCS) SWITCH

The AFCS switch is an autopilot emergency override switch on auxiliary flight control panel (figure 1-21), placarded AFCS, and has positions marked NORM and DISC ORIDE. The switch is guarded to NORM. In DISC ORIDE, signals from roll and autopilot commands, roll trim commands, and pitch damper trim inputs are removed from roll and pitch damper systems. The reference not engaged caution lamp is also illuminated.

AFCS RESET BUTTON

The AFCS reset button on left console (figure 1-5) is a momentary pushbutton switch marked AFCS RESET. The pushbutton is provided to allow ground checkout of dampers. However, it may be used in flight to verify malfunctions or clear malfunctions from extraneous sources. When the button is depressed, pitch, roll, and yaw damper lamps and their respective channel lamps on main caution panel will go out and dampers and their respective electronic channels will simultaneously reset to accept inputs for logic voting. If an erroneous input is present at time the button is released, appropriate indicator will illuminate.

GROUND CHECK PANEL

The ground check panel (figure 1-23) on the aft bulkhead provides switches and buttons for ground functional test and electrical power controls for the following: mach trim, engine fire detction system, engine inlet spikes, boarding ladder, stores refuel, instruments, gyros, flight control computers and rate gyros, damper servo, accelerometer, CADC system, and engine ignition. The description and function of each switch and button is described under the applicable system. A door is provided to cover ground check panel and shall be closed for in-flight operation. The switches on this panel are held in the proper in-flight position by the door being closed.

Computer Power Switches

The computer power switches, marked No. 1, No. 2, and No. 3, are on the flight control portion of ground check panel (figure 1-23). When any one of the switches is placed in UP, activating power is applied to the selected branch in each pitch, roll and yaw computer, and feel and trim assembly. The switches are held in this position when the door to the panel is closed.

Damper Servo Button

The damper servo button marked DAMPER SERVO is on the flight control portion of ground check panel (figure 1-23). When the damper servo, rate gyro channel B and channel C buttons, and flight control master test switch are depressed and held, electrical power to valve No. 1 on each damper servo is

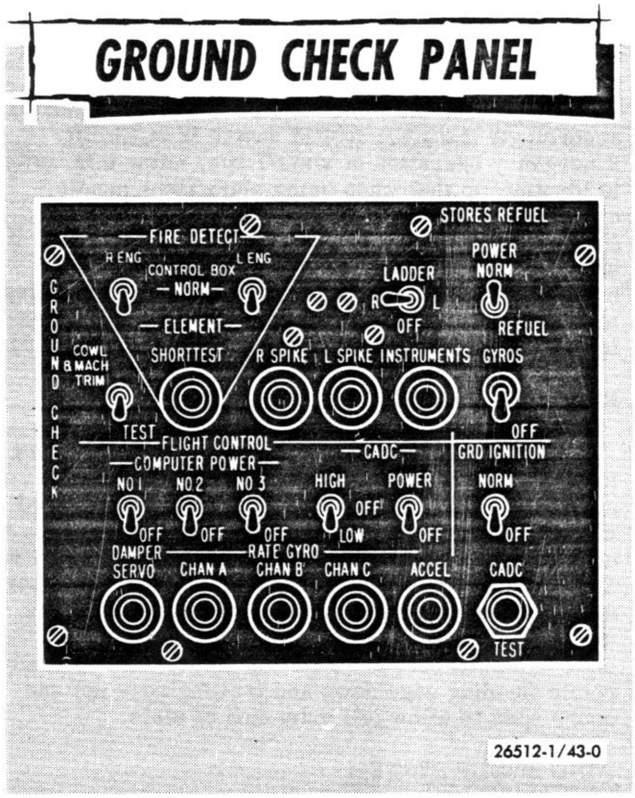

Figure 1-23

interrupted. This results in an electrical command signal from each computer, causing damper servos to vote hydraulically and the pitch, roll, and yaw damper and channel caution lamps to illuminate.

Rate Gyro Test Buttons

The rate gyro test buttons (CHAN A CHAN B and CHAN C) are on the flight control portion of ground check panel (figure 1-23). When two or more of the buttons are depressed, in conjunction with the flight control master test switch, respective rate gyros are torqued, resulting in a predetermined displacement of primary flight control surfaces. The CHAN A button, when depressed, torques "A" gyros in pitch, roll, and yaw channels. The CHAN B and CHAN C buttons, when depressed, torque their respective gyros. If only one channel button is pressed, no surface motion will take place and pitch, roll and yaw channel and pitch and roll gain changer caution lamps will illuminate.

MAXIMUM SAFE MACH ASSEMBLY

The maximum safe mach assembly (MSMA) receives mach number, pressure altitude and true air temperature signals from the central air data computer and wing sweep position from the wing sweep sensor, and provides outputs to the maximum safe mach indicator and to the reduce speed warning lamp. The MSMA computes the maximum continuous safe mach of aircraft regardless of whether the limitation is due to structural loading limitation of temperature, and provides a signal to the maximum safe mach indicator. It also continuously compares aircraft mach number to the maximum safe mach, computed as a function of structural load limit, and provides a signal to illuminate the reduce speed warning lamp when aircraft reaches maximum allowable speed. The MSMA requires 115-volt AC power from essential AC bus through the central air data computer power switch and 28-volt DC power from the essential DC bus.

SLATS AND FLAPS

SLATS

Each wing is equipped with a leading edge slat. Each slat is divided into five sections which are connected and operated as one unit. ~~The slats operate in conjunction with the main flaps and are connected to the main flap drive assembly by flexible drive shafts. On the extend cycle, the slats will extend to full down position before the main flaps start to extend.~~ On the retract cycle, the flaps will fully retract before slats start to retract. Asymmetrical slat travel is prevented by an asymmetry device which, when sensing asymmetrical slat travel, will close main flap drive control valve. Once the flap drive control valve has closed, slats and flaps cannot be extended or retracted by either normal or emergency mode.

Note

A mechanical interlock prevents the slats/flaps from being lowered when wing sweep angle is greater than 26 degrees.

FLAPS

The wing flaps are full span, multisection, Fowler-type flaps. Each wing flap is divided into five sections. The five outer sections, designated as main flaps, are mechanically connected and operated, as one unit. The inboard section, designated as the auxiliary flap, operates independently from the main flaps. The main flaps are powered by a single hydraulic motor which is connected to a gear box in the fuselage section. The hydraulic motor and gear box assembly drive a torque shaft which is connected through gear boxes to mechanical actuators attached to the flaps. An electric motor mounted on this same gear box provides an emergency mode of operation if a utility hydraulic system failure occurs. The auxiliary flap actuators are disabled when either wing sweep angle switch senses more than 16 degrees wing sweep or when the wing sweep handle is at a position greater than 16 degrees. Also, a mechanical interlock locks the slats/flaps handle in UP when either wing sweep angle is greater than 26 degrees or when wing sweep handle is at a position greater than 26 degrees. Asymmetrical flap travel is prevented by an asymmetrical sensor which signals the flap drive control valve to close. Once the flap drive control valve has been closed and torque shaft brakes are engaged by this method, flaps cannot be extended or retracted by either normal or emergency mode. Integral with each main flap section is a mechanically

controlled vane. As the flap extends downward, the vane is positioned by a mechanical linkage to provide proper airflow through the space between flap leading edge and spoiler trailing edge. The auxiliary (inboard) flaps are independently operated by electrical actuators and are energized only when 28 degrees or more flaps are selected by the slats/flaps handle. There is no mechanical connection between auxiliary flaps since there is no necessity to prevent asymmetrical operation. Utility hydraulic system pressure operates a slats/flaps hydraulic motor, and 115-volt AC power is used to energize auxiliary flap actuators.

SLATS/FLAPS HANDLE

The slats/flaps handle on the left console (figure 1-5) has three positions marked UP, SLAT DOWN, and FLAP DOWN. A detent is located at the SLAT DOWN position and another detent is at approximately 15 degrees flaps down. When the handle is moved from UP to any position in SLAT DOWN area, a mechanical linkage opens the flap drive control valve, directing hydraulic pressure to the flap drive motor. The flap drive assembly rotates the flexible shafts connected to the slat drive mechanism to position the rotating glove and to extend the slats to a position corresponding to handle position. Moving handle down to slat down detent will cause slats to indicate fully extended. When handle is moved from SLAT DOWN detent to FLAP DOWN area, flap drive assembly will rotate flexible shafts connected to main flap actuators, extending main flaps to a position corresponding to handle position. The slats and flaps drive assembly is so designed that it will not extend flaps until slats are fully extended. When handle is moved down to a position corresponding to 28 degrees or more of flaps, a contact closes providing electrical power to the auxiliary flap actuators. Full down position of the slats/flaps handle will provide 40 degrees of flap deflection. The retraction cycle sequence is just the opposite from the extension cycle. Moving handle from full FLAP DOWN to full UP will first cause flaps to retract and then slats to retract.

SLATS/FLAPS POSITION INDICATOR

The slats/flaps position indicator is a part of the wing sweep and slats/flaps position indicator on the pilot's instrument panel (figure 1-3). The indicator displays main flap position in degrees and slats and auxiliary flaps position in a window as either UP or DN (down). When the slats or auxiliary flaps are in transit or when electrical power is turned off, a barber pole is displayed in indicator window.

EMERGENCY SLATS/FLAPS SWITCHES

There are two switches placarded SLATS/FLAPS on the auxiliary flight control panel (figure 1-21) on the left console for emergency operation of the slats/flaps. One switch has two positions labeled EMER and NORM. The other switch has three positions labeled RETRACT, EXTEND, and spring-loaded to an unmarked off position. The EMER position of EMER/NORM switch disables flap hydraulic motor which will allow the emergency slats/flaps electric motor to operate when energized. When the switch is in NORM, slats/flaps are controlled by slats/flaps handle. The RETRACT/EXTEND switch controls slats/flaps electric motor and positions slats/flaps accordingly if EMER/NORM switch is in EMER. Emergency operation of slats/flaps, using this switch is identical to that when using slats/flaps handle, except that electric power is used to operate slats/flaps drive instead of hydraulic power. However, it should be noted that EMER/NORM switch does not control auxiliary flaps since they are only controlled by slats and flap handle. Emergency slats/flaps extension or retraction takes approximately 60 seconds.

ROTATING GLOVES

The outboard edges of wing gloves, adjacent to wing inboard leading edges, are equipped with movable surfaces to allow full forward movement of inboard slats. These surfaces are called rotating gloves. A door forms the lower surface of each rotating glove. Each rotating glove and its associated door are operated by a mechanical actuator and linkage which is connected to the slats drive flexible shaft. When the slats are extended, the rotating gloves automatically rotate (leading edge down and trailing edge up) and doors open to allow full extension of slats.

WING SWEEP SYSTEM

The variable sweep wings (figure 1-24) are moved to and held in position by two hydraulic, motor-driven, linear actuators. Range of wing sweep is from 16 to 72.5 degrees. The actuators are mechanically interconnected to insure positive synchronization. The right actuator is furnished power by the primary hydraulic system, and left actuator is furnished power by the utility hydraulic system. Should either hydraulic system fail, the load transfer capability of the mechanical interconnect will still provide wing actuation for both wings by the remaining system. However, actuation under this condition will be at a reduced rate commensurate with actuator loading. Wing position is controlled by a closed loop mechanical servo system in response to an input signal from the wing sweep handle. The maximum rate at which wings extend or retract is controlled by flow-limiting devices in hydraulic lines. Directional reversal, due to aerodynamic loads, is prevented by nonreversing (acme-type) threads in actuator. The wing sweep handle is locked in the 16-degree position by a solenoid-operated latch whenever auxiliary flaps are out of zero position. Also, a mechanical interlock prevents wing sweep handle from being moved past the 26-degree position when either slats/flaps handle is out of UP or main slats or flaps are out of fully the retracted position.

WING SWEEP CONTROL HANDLE

The wing sweep control handle (figure 1-24) is shaped like a pistol grip and is spring-loaded to a stowed position under the canopy sill on left side of crew module. A lock is provided to prevent inadvertent movement of wing sweep control handle while in stowed position.

WING SWEEP AND PYLON SYSTEM

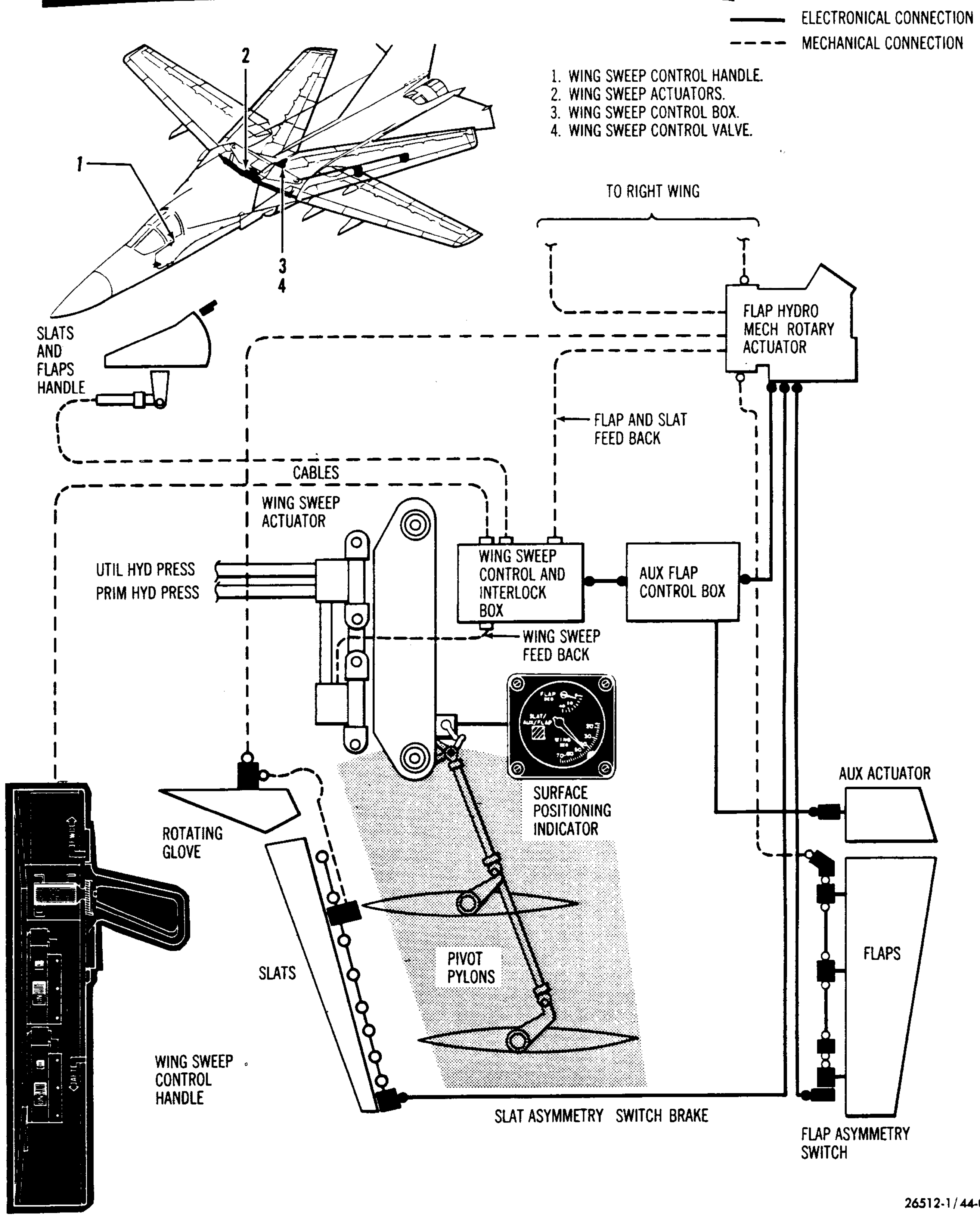

Figure 1-24

To adjust wing sweep, handle must be rotated to vertical position to unlock it; then it can be moved forward or aft as necessary. The handle is mechanically linked to wing sweep control valve. The handle is pulled aft to sweep the wings aft and pushed forward to sweep the wing forward.

Wing Sweep Handle Lockout Controls

Two wing sweep handle lockout controls, one labeled FIXED STORES and the other labeled WEAPONS, are just above and aft of wing sweep control handle. When either control is moved forward, word ON is visible, and a latch extends which prevents aft movement of wing sweep handle past latch. When either control is moved aft, word OFF is visible and latch retracts. The fixed stores lockout control, when ON prevents wing sweep handle from being moved aft past the 26-degree position. This is the sweep angle at which fixed pylons and stores are in a streamlined configuration. The weapons lockout control, when ON, prevents wing sweep handle from being moved further aft past the 55-degree position. Certain weapons, when mounted on inboard pivot pylons would strike fuselage, if wings were to be swept further aft beyond this point. The wing sweep handle lockout controls only prevent aft movement of wing sweep handle. Forward movement is never restricted.

Wing Sweep Handle 26-Degree Forward Gate

A wing sweep handle 26-degree forward gate above the wing sweep handle is provided to stop forward motion of wing sweep handle at 26 degrees. The gate is thumb-actuated and is spring-loaded to the latched position. Depressing the gate will retract a latch, allowing wing sweep handle to be moved forward past the 26-degree position.

WING SWEEP POSITION INDICATOR

The wing sweep position indicator is a part of the surface position indicator on the pilot's instrument panel (figure 1-3). The indicator displays the wing position in degrees and is graduated in 2-degree increments from 16 to 72 degrees. The angle of wing sweep is monitored by a transmitter which mechanically follows change in wing position and converts this information to an electrical signal which drives the wing sweep indicator.

SPEED BRAKE

The speed brake, which also serves as the main landing gear forward door, is provided as an aid to deceleration during flight. The speed brake is hydraulically operated and may be used as a speed brake only when landing gear is up and locked.

SPEED BRAKE SWITCH

A three-positioned speed brake switch placarded IN, OFF, and OUT is on right throttle. The switch is thumb-actuated and slides forward to retract (IN) and aft to extend (OUT), and is spring-loaded from OUT to a center OFF detent. It is also detented in the IN position. The electrical circuit is activated when the landing gear is up and locked and the speed brake door is fully retracted to enable in-flight operation of the speed brake. It can be held in any position between fully IN and fully OUT while in-flight with landing gear up and locked by moving switch to OFF during brake door transit.

SPEED BRAKE INDICATOR

A speed brake indicator on left side of pilot's instrument panel (figure 1-3) shows operating positions of speed brake as integrated with landing gear. There are four different indications, UP, 100%, dotted area, and a barber-pole that will appear to indicate speed brake positions. When the landing gear is up, indicator will display the following: UP, when the speed brake is fully retracted; 100%, when speed brake is fully extended; and a dotted area, when the speed brake is transitioning or is stopped in any position other than fully retracted or fully extended; the barber-pole appears when there is no power to the instrument. When the landing gear is down, the indicator will display the following: UP, when the speed brake is in the trail position; and the dotted area, when the speed brake is transitioning to the trail position. The 100% display will momentarily show while the speed brake is in the fully extended position during extension of the main landing gear, the barber-pole appears when there is no power to the instrument.

GROUND ROLL SPOILERS

Deceleration during ground roll is aided by symmetrical extension of flight control spoilers which reduce aerodynamic lift and allows maximum effectiveness of the wheel brakes.

GROUND ROLL SPOILER SWITCH

The ground roll spoiler switch on the left side-wall (figure 1-5) has two positions - BRAKE and OFF. If weight of aircraft is on landing gear and both throttles are in IDLE, positioning this switch to BRAKE will cause flight control spoilers to extend. Under the same conditions, placing switch to OFF will retract all spoilers. The spoilers cannot be extended with the spoiler switch when aircraft is in the air or when either throttle is not in IDLE.

LANDING GEAR SYSTEM

The landing gear is tricycle-type, forward retracting, and hydraulically operated. The main landing gear consists of a single common trunnion upon which two wheels are singly mounted. This arrangement of the main gear provides symmetrical main gear operation. The conventional nose landing gear pneudraulic strut has dual-mounted wheels and incorporates a hydraulically operated steer damper for nose wheel steering. The landing gear system is normally powered by the utility hydraulic system and an emergency pneumatic system is provided as an alternate means of extending the gear. The nose gear retracts into a nose wheel well, and the main gear retracts into a fuselage well.

MAIN LANDING GEAR

Two double-acting cylinders, one an uplock and one a downlock, function to lock the main landing gear in its respective positions. A single-acting cylinder retracts the gear to the up position, and free fall provisions allow it to extend to the down position. There are two, an aft and a forward, main landing gear doors. The forward door, which also serves as the speed brake, is controlled hydraulically in sequence with landing gear retraction and extension. The aft door is mechanically linked to the main gear for proper sequencing, and a mechanical connection between the main gear and the speed brake selector valve establishes proper sequencing of the forward door. Two weight on wheels switches, on the lateral trunnion beam, prevent normal gear retraction while the aircraft is on the ground.

NOSE GEAR

Three hydraulic actuators are provided for the operation of the nose landing gear and nose wheel well doors. A single-acting actuator retracts nose landing gear. An uplock actuator locks nose landing gear in retracted position and also, through linkages, opens and closes the nose wheel well doors. A downlock actuator locks the nose landing gear drag strut down when the nose landing gear is extended.

LANDING GEAR PANEL

The landing gear panel (figure 1-25) on the forward end of the left console provides for the landing gear handle, the landing gear handle lock release button, null trim lamp, utility hydraulic system isolation switch, and auxiliary brake handle. The description and function of null trim lamp, utility hydraulic system isolation switch, and auxiliary brake handle are described under the applicable system.

Landing Gear Handle

The landing gear handle on the landing gear panel (figure 1-25) has two positions marked UP and DN. The handle has a gear unsafe warning lamp in the grip. Moving the handle to UP or DN will cause the following actions to occur.

Gear Up

When the handle is moved to UP, an electrical signal actuates a solenoid-powered valve, sending hydraulic pressure to nose gear downlock actuator, nose gear retract actuator, nose gear uplock door actuator, and speed brake door actuator. The nose gear unlocks and retracts. When it is almost retracted, it mechanically triggers nose gear uplatch which locks gear up and then closes and locks doors. The speed brake door (forward door) is extended and when it is sufficiently open, a linkage from door opens a valve which sends hydraulic pressure to main gear downlock actuator, main gear retract actuator, and uplock actuator. The gear then unlocks and retracts. When it is almost retracted, it mechanically triggers uplatch which locks gear up and also actuates a valve to close speed brake door. The aft door is mechanically sequenced closed and locked.

LANDING GEAR PANEL

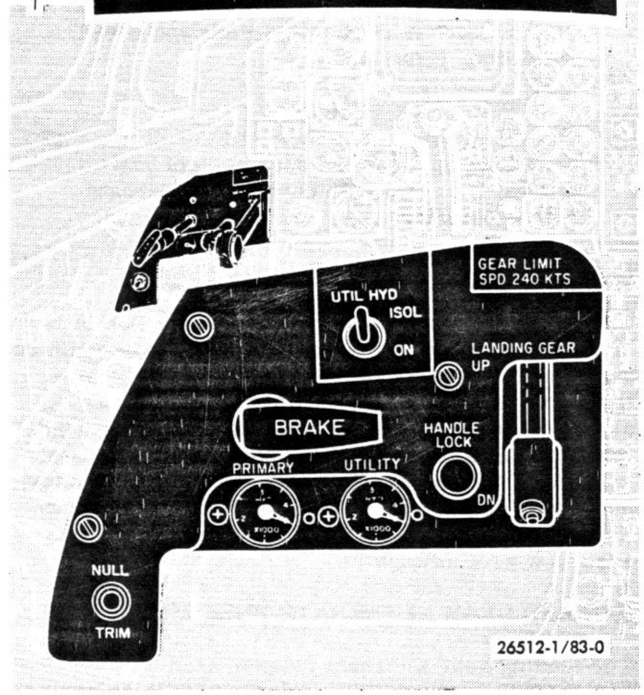

Figure 1-25

Gear Down

When the handle is moved to the DN position, an electrical signal actuates a solenoid-powered valve, sending hydraulic pressure to the nose gear uplock actuator, nose gear downlock actuator, and speed brake door actuator. The nose gear uplock actuator unlocks and drives nose gear doors open and locked, at which time the nose gear is allowed to free fall (extend) against snubbing of its retract actuator. When gear is almost extended, the downlock actuator drives it fully extended and locked. The speed brake door extends until adequate main gear clearance is obtained and then a mechanical linkage actuates a valve which pressurizes the main gear uplock actuator and downlock actuator. The uplock opens, allowing the gear to free fall (extend) against the damping of its retract actuator. When gear is extended, downlock actuates and speed brake door retracts to a trail position. The landing gear handle is locked in DN when weight of aircraft is on landing gear. A weight on wheels switch breaks an electrical circuit to a spring-loaded solenoid, permitting it to extend a mechanical lock which holds landing gear handle in DN. When weight of aircraft is removed from landing gear, weight on wheels switch closes electrical circuit and energized solenoid retracts lock freeing landing gear handle.

CAUTION

Do not reposition landing gear handle while landing gear is in transit. If a speed brake control valve malfunctions, gear and speed brake door could jam.

Note

A sticking speed brake door control valve will not interfere with normal or alternate gear extension.

Landing Gear Handle Lock Release Button

The landing gear HANDLE LOCK release button is on landing gear panel (figure 1-25). The button must be depressed to release the landing gear handle from UP position to lower gear. Normally, it is not necessary to depress button when retracting gear since gear handle is locked in down position by a solenoid which will release handle as weight of aircraft comes off wheels on takeoff. Should the solenoid malfunction, depressing the button will release the handle and allow gear retraction.

LANDING GEAR ALTERNATE RELEASE HANDLE

The landing gear alternate release handle on forward end of center console (figure 1-6) permits extension of landing gear if utility hydraulic pressure is not available. When handle is pulled, pneumatic pressure is directed to simultaneously open speed brake door and unlock nose and main gear uplock. The gear will then free fall to extended position. Pneumatic pressure will actuate nose and main gear down locks and retract speed brake door to trail position. Once gear has been extended by the alternate method, it cannot be retracted. If landing gear handle warning lamp remains illuminated after gear is extended and locked, indicating failure of the speed brake to return to the trail position, push the handle back in. This will allow the air load to push speed brake door to trail position.

CAUTION

As aircraft slows after landing, weight of door and lack of air load will allow door to extend and drag the ground. Stopping aircraft as soon as possible will prevent extensive damage to door. Any time this handle is pulled, it must be pushed back in before removing electrical power from aircraft. Otherwise, pressure in speed brake door actuator will extend door, causing damage from ground contact.

LANDING GEAR WARNING AND POSITION INDICATORS

Wheels Warning Lamp

The wheels warning lamp is to the right of the approach indexer on left glare shield (figure 1-3). It is a red light that will flash at 160 cycles per minute any time throttles are retarded to less than cruise (40 degrees) while flaps are in a position other than fully retracted, and one of following conditions exist: landing gear is not down and locked; speed brake is not in a trail position, or both.

Landing Gear Warning Lamp

The landing gear handle has a gear unsafe warning lamp contained in handle grip which will illuminate if any of the conditions are as follows:

Landing gear in transit
Landing gear down but not locked
Landing gear up but not locked
Landing gear down and locked but speed brake not in trail

Landing Gear Position Indicators

The integrated position indicators on left side of pilot's instrument panel (figure 1-3) shows the position of the nose and main wheels. Transition of the gear during retraction and extension is shown in the windows as a barber-pole indication. When landing gear is down and locked, a picture of wheels appears in appropriate windows and when gear is up and locked, UP is indicated.

NOSE WHEEL STEERING

The nose wheel steering aids directional control of aircraft while taxiing, and during takeoff and landing. In addition, it prevents nose wheel swivelling during roll-back after an arrested landing. The system is electrically engaged, hydraulically actuated and controlled by rudder pedals. When electrically engaged, steering signals from rudder pedal movement are transmitted to a hydraulic steering control which directs utility hydraulic pressure to the steering actuator. Nose wheel steering range is 40 degrees either side of center, with a free swivelling range of 360 degrees for towing or turning using differential braking. The steering mechanism also incorporates shimmy damping capability when not in use as a steering actuator. Hydraulic system pressure is not required for shimmy damping.

NOSE WHEEL STEERING SWITCH

Nose wheel steering is selected by a press and hold switch on the control stick grip (figure 1-20) when weight of aircraft is on landing gear. Nose wheel steering is automatically engaged when arresting hook is down and full aircraft weight is on landing gear.

Note

Nose wheel steering will not be available if landing gear is extended using alternate release handle.

TAIL BUMPER SYSTEM

The tail bumper protects the control surfaces, engines and portions of the airframe from damage if tail inadvertently contacts the ground during ground

handling. The tail bumper also provides limited protection during overrotation on takeoff and during landings. In flight, the tail bumper is held in fully retracted position by hydraulic pressure in the tail bumper lift cylinder. The hydraulic pressure is ported to tail bumper lift cylinder from speed brake control valve. When landing gear is extended and speed brake returns to trail position, lift cylinder pressure is relieved and tail bumper is extended by pneumatic action of tail bumper dashpot. The dashpot, which functions as impact shock absorber, has its own separate reservoir that is charged with compressed nitrogen. Retraction of landing gear allows hydraulic pressure to be again ported to tail bumper lift cylinder to retract bumper and hold it in this position.

Note

Alternate extension of landing gear will not extend tail bumper, but will allow it to extend by spring action of dashpot.

TAIL BUMPER INDICATOR

A tail bumper indicator on left side of pilot's instrument panel (figure 1-3) shows the position of tail bumper. The word UP appears in indicator window when tail bumper is up. A blank indication will appear when tail bumper is down. A barber-pole condition shows when bumper is either extending or retracting.

LAUNCH BAR

The launch bar is mounted on nose gear caster barrel and shock strut. Basically, it is a hands-off operation with the exception of one manual external function, which is to attach the release element to the trail bar and then attach this assembly to the aircraft.

LAUNCH BAR SWITCH

A launch bar switch on the left console (figure 1-5), placarded LCH BAR, has three positions marked EMERG UP, UP, and DOWN. When DOWN is selected, the following conditions must be met for the extend solenoid on control valve to be energized. These are: aircraft must be on the ground; throttles must be at a position of less than cruise; and nosewheel must be in the steering range. When the extend solenoid is energized, utility hydraulic pressure is ported to extend side of launch bar actuator which drives the lowering cam down into contact with the uplock mechanism, releasing uplock and forcing launch bar to down position. When either throttle is advanced to cruise position, switch is automatically returned to UP position, and launch bar will retract after catapult launch. The EMERG UP position of launch bar switch is used to force control valve to up position which ports hydraulic pressure in launch bar actuator to return allowing spring force to bring launch bar to stowed position.

Note

If launch bar does not lock up after a catapult launch, landing gear handle will remain locked in down position by solenoid lock.

LAUNCH BAR LAMP

The launch bar lamp on pilot's instrument panel displays the letters LAUNCH BAR in red when illuminated. It will be illuminated when conditions exist; as follows: aircraft is airborne with landing gear down and locked but launch bar is not in the stowed position; the aircraft is on the ground and launch bar actuator is in a position other than up.

BRAKE SYSTEM

Each main landing gear wheel is equipped with a hydraulically operated multiple disc brake. Pressure for operation of the brakes is supplied by the utility hydraulic system for normal operation and by two hydraulic accumulators whenever utility hydraulic pressure is not available. Anti-skid control, automatic braking during landing gear retraction, and an auxiliary brake are provided. Normal brake operation is controlled by conventional brake pedals, each mechanically connected to brake metering valves. The brake hydraulic system is a dual-normal type, separated into two circuits. Each circuit receives pressure from the utility hydraulic system, but each circuit operates independently of the other. One circuit operates one half of the pressure pistons on left brake and one half the pressure pistons on right brake. The other circuit operates the other half of pistons on each brake. During normal operation of brakes, pressure is metered to brakes from both hydraulic circuits in proportion to applied force on brake pedals. If one hydraulic circuit becomes inoperative, the brake system can provide increased pressure to the remaining operative circuit, thereby permitting approximately the same braking effectiveness as is normally available with both circuits operative.

Greater than normal brake pedal travel and slightly higher pedal-force are required to achieve increased pressure to a single hydraulic circuit. The dual-normal type brake hydraulic system provides emergency brake operation automatically; therefore, actuation of an emergency brake control handle is not required. Two hydraulic accumulators are provided in the system to supply brake system pressure if failure of the utility hydraulic system occurs. Each accumulator is precharged and supplies pressure to only one of the individual brake circuits. Fully charged accumulators will provide 18 full-pressure brake applications and three emergency brake applications. A priority valve, which limits the quantity of fluid which can be displaced from brake accumulator through brake metering valves by actuating brake pedals, is included in each hydraulic circuit. If brake accumulators are not replenished as fluid is displaced by repetitive brake applications or by anti-skid cycling, priority valves will close when accumulator pressure has been reduced to approximately

1000 psi. When accumulator pressure is 1000 psi, sufficient fluid volume for five brake applications is remaining. After priority valves close, remaining fluid can be utilized only by pulling auxiliary brake handle. No braking action can be achieved by actuating brake pedals after pulling auxiliary handle when less than 1000 psi is in the accumulators.

CAUTION

Do not actuate brake pedals in flight. When utility hydraulic pressure is isolated from the brake system, there is no way to replenish brake accumulators. If the utility hydraulic system fails after brake accumulators are bled off to below 1000 psi, there will be no braking available with brake pedals on landing.

ANTI-SKID

Anti-Skid control is provided for normal braking. Solenoid-operated valves in each brake and anti-skid control valve assembly function to release brake pressure in response to electrical signal received from the anti-skid control system, as impending wheel skids are detected. The solenoid valves will reapply brake pressure upon being de-energized after wheel returns to normal speed.

Anti-Skid Switch

A two-positioned toggle switch with positions ANTI-SKID and OFF is on left console. Placing switch in ANTI-SKID will provide anti-skid control during normal and emergency braking. With switch in OFF, anti-skid control will not be available and brake pressure will be in direct response to pedal pressure.

AUXILIARY BRAKE HANDLE

An auxiliary brake handle marked AUX BRAKE is on the landing gear panel (figure 1-25). When handle is pulled, a mechanical linkage opens a selector valve admitting pressure from hydraulic accumulators directly into brake lines downstream of brake control valve. The primary function of the auxiliary brake control handle is to apply brakes while aircraft is parked, and it can be used to set brakes for engine runup. A secondary function of auxiliary brake control is to serve as a supplemental emergency brake if brake accumulator pressure is reduced sufficiently to prevent normal brake application by pedal actuation. Brake pressure cannot be metered by auxiliary brake handle. The total accumulator pressure is ported directly to brake cylinders. Therefore, auxiliary brake handle should not be pulled while aircraft is in motion, except when braking cannot be achieved by pedal actuation. No anti-skid action is available when auxiliary handle is pulled.

CAUTION

Pulling the auxiliary brake handle while aircraft is moving will cause wheels to lock and result in tire skidding or blowout.

ARRESTING HOOK SYSTEM

The arresting hook system consists of an arresting hook, hook dashpot, retract actuator, uplatch actuator, and a solenoid operated control valve. The up or down position of arresting hook is normally selected by a two-positioned, toggle switch. An emergency handle connected through mechanical linkage to the uplatch mechanism provides for emergency hook extension. The arresting hook is retracted by hydraulic pressure from the utility system and extended by the dashpot.

ARRESTING HOOK SWITCH

The arresting hook switch on the left side of MCO's instrument panel (figure 1-4) is a two-positioned toggle switch placarded HOOK. The positions are marked UP and DOWN. When switch is DOWN, a control valve is energized which ports hydraulic pressure to release actuator and ports retract actuator to return. This releases the uplatch, allowing dashpot to lower arresting hook. The up position of switch reverses extension cycle. The uplatch actuator is spring-loaded to latched position.

ARRESTING HOOK HANDLE

The arresting hook handle on the left side of MCO's instrument panel (figure 1-4) placarded EMER ARG HOOK is connected to a low friction, push-pull type mechanism. The mechanism provides a direct mechanical linkage to arresting hook uplatch mechanism and control valve in tail cone. The arresting hook is released by grasping handle and pulling aft for approximately four inches. Approximately one second is required for arresting hook to extend. The hook cannot be raised to its stowed position from cockpit after being extended by pulling the arresting hook handle.

ARRESTING HOOK CAUTION LAMP

The arresting hook lamp adjacent to the arresting hook switch is a disagreement lamp. When the hook is in a position other than the position selected by the switch, the lamp illuminates displaying the letters HOOK in amber.

FLIGHT INSTRUMENTS

Optimum control during takeoff, climb, cruise, attack, letdown, and landing is provided by the Vertical Display Indicator Group. Refer to Section VIII for further information.

ATTITUDE DIRECTOR INDICATOR

The attitude director indicator on pilot's instrument panel (figure 1-3) provides backup attitude information if Vertical Display Indicator Group fails. The indicator displays pitch and roll information on an attitude sphere in relation to a minature aircraft. Pitch and roll signals are received from auxiliary flight reference system (AFRS). The indicator receives 115-volt AC power from essential AC bus. An OFF

warning flag will appear on lower left face of the indicator, if power fails or AFRS has a malfunction. The indicator has a vertical azimuth bar and a horizontal glide slope bar positioned in front of the miniature aircraft, and a glide slope (G/S) alarm flag. The vertical bar and horizontal bar will come into view when the instrument is receiving signals from a ground station (Tactical Data System) through Data Link when the mode selector switch is in Automatic Carrier Landing (ACL) mode. A pitch trim knob on lower right side of instrument is used to adjust horizon line to correct for changes of aircraft pitch attitude.

HORIZONTAL SITUATION INDICATOR

The horizontal situation indicator (HSI) (figure 1-26) on the pilot's instrument panel displays heading, course, bearing and distance information. The rotating compass card is read against a fixed lubber line and receives magnetic heading signals directly from inertial navigation system or auxiliary flight reference system.

The heading marker at the periphery of compass card is positioned automatically or manually. It is positioned automatically by a signal from data link (DL) or inertial navigation system (INS). It is positioned manually by the HDG SET selector, on the power left corner of the indicator. The heading marker will indicate selected heading and will rotate with the compass card.

The course pointer which is read against the compass card is positioned automatically or manually. It is positioned automatically by a signal from INS, and manually by CRS SET selector, on lower corner of indicator. The course bar (center segment of the course pointer) indicates aircraft steering information for the mode selected. The deviation is shown in relation to fixed miniature aircraft symbol. The degree of deviation is measured by four course dots positioned perpendicular to course bar. A course bar alarm flag shows NAV in a display window whenever the course bar indicates an invalid deviation or there is a loss of electrical power. Selected course is also displayed by COURSE counter in the upper right corner of indicator.

Bearing pointer No. 1 and bearing pointer No. 2 are on periphery of compass card and provide magnetic bearing information to a TACAN station, ADF station or target. Bearing pointer No. 1 receives its signals from TACAN system or airborne missile control system (AMCS). Bearing pointer No. 2 receives its signal from UHF ADF system.

The "to-from" arrow indicates whether the course selected, if intercepted and flown, will take aircraft toward or away from TACAN station.

The distance counter, placarded MILES is in upper left corner of indicator. It indicates distance to an INS computed destination, range of AMCS target, or distance to a TACAN station in nautical miles. A warning flag covers the MILES counter when it is not in use or an unreliable signal is received.

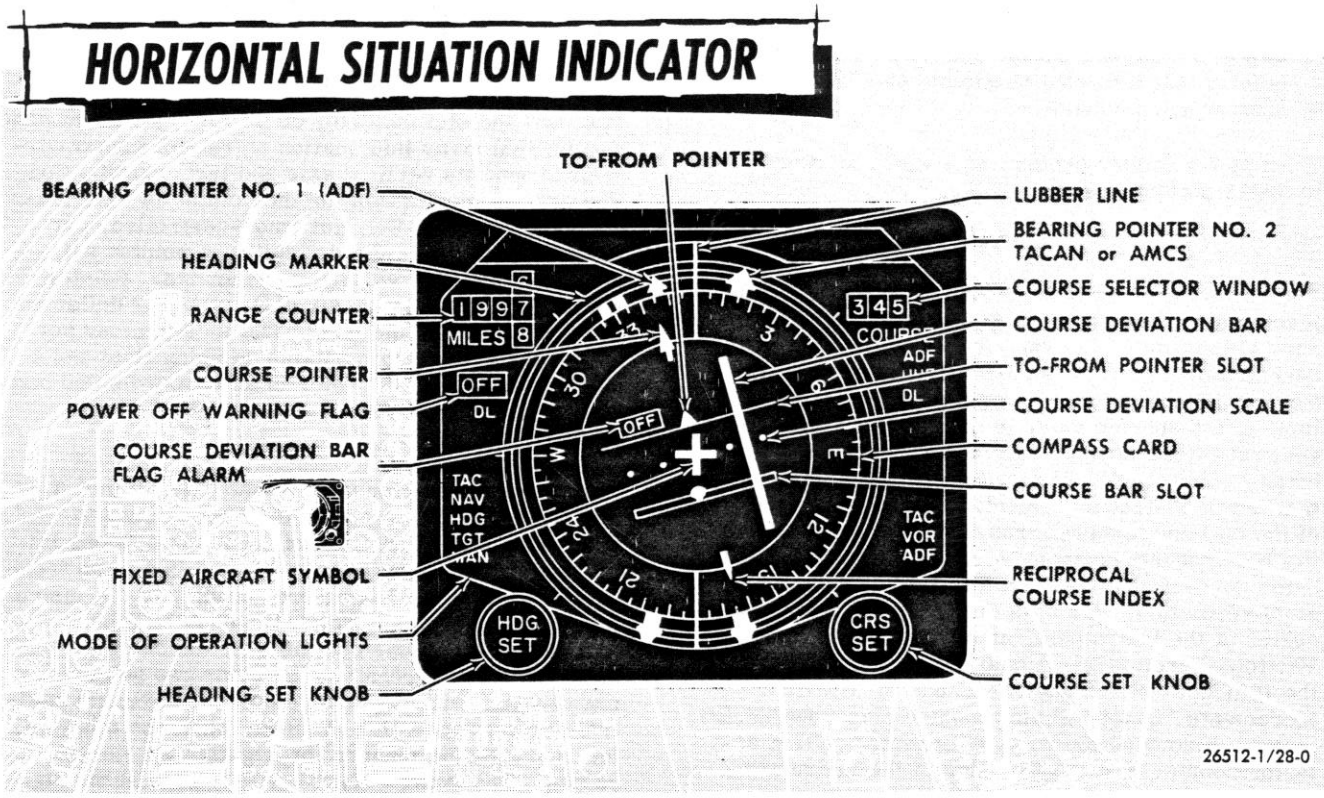

Figure 1-26

Mode lights operate in relation to mode selected on display control panel (figure 1-5). Loss of power to HSI will cause an OFF warning flag to appear below MILES counter. The HSI operates on 115-volt AC power from essential AC bus.

AIRSPEED/MACH NUMBER INDICATOR

The airspeed/mach number indicator on left side of pilot's instrument panel (figure 1-3) provides values of airspeed, mach number, command airspeed, command mach number, and maximum safe mach number on a single presentation. Pneumatic pressures from pitot-static system activate airspeed and mach mechanisms in indicator. Command mach or command airspeed signals are provided by the digital data link (DL) system. The vertical display indicator group (VDIG) is provided with a command scale change from the airspeed/mach number indicator. The range and derivation of each indicator display are as follows:

INDICATED AIRSPEED - 80 to 850 knots, is obtained from pitot-static-operated airspeed mechanism that drives airspeed pointer.

MACH NUMBER - 0.4 to 2.8 mach, are derived from static-operated altitude mechanism that drives a moving scale (subdial) to indicate mach number against airspeed pointer.

COMMAND AIRSPEED - 80 to 850 knots, and command mach, 0.4 to 2.8 mach, indications are obtained from a servo-driven index marker that presents mach against airspeed (mach dial).

MAXIMUM SAFE MACH - at altitudes of 1,000 to 80,000 feet are obtained from a servo-driven pointer that indicates maximum safe mach against mach subdial.

When power failure occurs, safe mach marker drives to the 12 o'clock position.

SERVO BAROMETRIC ALTIMETER

The servo barometric altimeter on left side of pilot's instrument panel (figure 1-3) displays accurate pressure altitude in feet. The altimeter is electrically operated by a synchro signal received from CADC for normal operation. An integral standby mechanism, of the aneroid type, is incorporated and presents pressure altitude with normal barometric setting correction. The face of the altimeter is marked in 50-foot increments around the periphery of the dial and numerals ranging from 1 to 10 indicate the 100-foot increments. A single-dial pointer indicates correct altitude from 0 to 1000 feet in a complete revolution. A single drum-type counter, in the center of the instrument, also shows altitude in 100-foot increments. A dual digital counter to left of the 100-foot counter shows altitude in 1000-foot increments. A four digital counter placarded IN. HG. is used in conjunction with an altimeter setting selector control on lower left side of the instrument.

A selector level is on lower right corner of indicator and marked RESET/STBY. Normally the standby mode will be in the center position and the altimeter will receive its electrical inputs from central air data computer. However, if this system becomes unreliable, the altimeter will receive its inputs automatically and directly from the pitot static system. A failure warning flag marked STBY will appear on the dial face, indicating altimeter is in mechanical or standby mode of operation. The STBY flag will be hidden when the instrument is in normal electrical mode of operation. The electrical mode of operation may be selected by manually positioning the standby mechanism to RESET. Conversely, standby mode of operation may be selected by positioning the selector lever to STBY.

VERTICAL VELOCITY INDICATOR

The vertical velocity indicator on left side of pilot's instrument panel (figure 1-3) is contained within a sealed case, and connected to a static pressure line through a calibrated leak. The reaction of a diaphragm inside the case to changing pressure is represented on face of indicator by a linkage system of gears and levers. The instrument will automatically compensate for changes in temperature. The immediate response of the diaphragm to atmospheric pressure is measured against trapped static pressure inside the case. When climbing or descending at a constant rate, a definite ratio between the diaphragm pressure and the case pressure is maintained through the calibrated leak. When aircraft is leveled, calibrated leak requires approximately 6 to 9 seconds to equalize two pressures, causing a lag in proper reading. When establishing a climb or descent, this lag is again apparent. Sudden or abrupt changes in attitude may cause erroneous indications due to sudden change of air flow over static probe.

TURN AND SLIP INDICATOR

The turn and slip indicator on pilot's instrument panel (figure 1-3) gives information on rate of turn of aircraft around its vertical axis and turn coordination. The driving mechanism for the pointer is a permanent-magnet type, DC, governor-controlled, gyromotor. A needle-width deflection of pointer will produce a 360-degree turn in 4 minutes. Pointer motion is damped by an air dashpot and is deflected in the direction of the turn. The inclinometer portion of the instrument contains damping fluid and a ball that moves from center in an uncoordinated turn.

ACCELEROMETER

The accelerometer on pilot's instrument panel (figure 1-3) is a direct reading instrument used to measure the accelerations of the aircraft along its vertical axis. The dial is graduated in g units from -5 g's to +10 g's. The normal reading of the instrument at rest is +1 g. The instrument has three pointers of which one continuously indicates vertical acceleration of aircraft. The other two pointers, one of which will stop and remain at maximum positive acceleration value attained, while the other will function in same manner for negative acceleration values. These two pointers will remain at the highest values reached until reset by depressing a knob on lower left corner of instrument.

ANGLE-OF-ATTACK SYSTEM

The angle-of-attack system measures the angle between the longitudinal axis of the aircraft and the relative wind. The indicating system provides an indication of angular position of wing chord in relation to aircraft flight path. This indication is used for approach monitoring and to warn an approaching stall. The system includes a vane-type transmitter, indicator, and an indexer. The indexer and the indicator are electrically slaved to a sensor vane transmitter. In flight, the vane which is on left side of the fuselage will align itself with the airflow. Rotation of the vane generates an indicated angle-of-attack signal to central air data computer. The central air data computer converts a signal to true wing angle-of-attack and sends this signal to angle-of-attack indicator and also illuminates the angle-of-attack indexer. A damper assembly prevents rotational overshoot and flutter of the vane due to turbulence. Vane anti-icing is provided by means of a 115-volt AC heating element in leading edge of the vane. The heating element is controlled by probe heater switch on external environment panel (figure 1-34).

Angle-of-Attack Indicator

The angle-of-attack indicator on left side of pilot's instrument panel (figure 1-3) has a pointer that is driven by a servo motor and rotates over a card graduated from 0 to 30 units. Optimum angle-of-attack for landing approach is marked by a lighted area at the three o'clock position on instrument. The units do not reflect angle-of-attack in degrees. A readout window on the instrument face will indicate OFF when there is no power to the servo motor.

Approach Indexer

The approach indexer on left glare shield (figure 1-3) has two arrows and a circle illuminated by lamps to provide approach information. The arrows are positioned vertically with the circle located between the two. The cam-operated switches in the angle-of-attack indicator also control the approach indexer. The upper arrow is for high angle-of-attack, the lower arrow is for low angle-of-attack, and the circle is for optimum angle-of-attack. When both arrow and circle appear, an intermediate position is indicated. The indexer lamps function only when landing gear is down. A flasher unit causes indexer symbol lamps to pulsate when an unsafe angle-of-attack exists.

TOTAL TEMPERATURE INDICATOR

The total temperature indicator on right-hand wing panel (figure 1-4) displays temperature of the air sensed by total temperature probe. The indicator is an electrical resistance-type instrument that uses a remote temperature sensing probe, an amplifier and a motor to position the indicator pointer. The total temperature sensing probe is equipped with a heating element for anti-icing. The face of the indicator is graduated in 10-degree increments from -50°C to +250°C, with a critical temperature index mark of 153.3°C and a maximum temperature index mark at 214.3°C. A digital readout counter in the face of indicator, marked SEC TO GO, indicates time remaining for operation in the critical temperature range between 153.3°C and 214.3°C. The counter will start to drive down from 300 seconds toward zero and an amber total temperature caution lamp will light when critical temperature of 153.3°C is reached. The counter will continue to drive until one or more of the following conditions are met: until it reaches zero; until temperature is reduced below 153.3°C; or until maximum temperature index of 214.3°C is reached. When the maximum temperature index is reached or when the counter is driven to zero, a red reduce speed lamp will illuminate. The counter will reverse and drive back to 300 seconds any time temperature falls and remains below 153.3°C. If reduce speed warning lamp is illuminated, it will go out as counter starts to drive back. The total temperature caution lamp will go out when counter has driven back to 300 seconds. An OFF flag will appear in face of indicator when power is removed from instrument. The indicator operates on 115-volt AC power from essential AC bus. A pushbutton test switch placarded INSTRUMENTS on ground check panel (figure 1-23) is for functionally testing total temperature indicator. Depressing button will cause pointer to drive up-scale past maximum temperature index, start timer, illuminate reduce speed warning lamp, and actuate indicator's off flag.

Critical Temperature Caution Lamp

The critical temperature caution lamp on main caution lamp panel (figure 1-28) will illuminate any time aircraft is operated above critical temperature of 153.3°C. When illuminated, words CRITICAL TEMP appear on lamp face. Once illuminated, lamp will remain illuminated until total temperature counter has reversed and driven back to 300 seconds.

Reduce Speed Warning Lamp

The reduce speed warning lamp on pilot's instrument panel (figure 1-3) functions in conjunction with total temperature indicator to indicate that aircraft has flown for at least 300 seconds in the critical temperature range of from 153.3°C to 214.3°C, or that maximum temperature index of 214.3°C has been reached or exceeded. When illuminated, words REDUCE SPEED are visible in red on face of lamp. If lamp was illuminated due to expiration of 300 seconds in the critical temperature range, it will remain on until temperature is reduced to below 153.3°C and total temperature counter has reversed and started to drive back to 300 seconds. If the lamp was illuminated upon reaching maximum temperature index of 214.3°C as the counter was driving to zero, it will go out as soon as temperature is reduced below 214.3°C.

CENTRAL AIR DATA COMPUTER (CADC)

The central air data computer system gathers and processes aerodynamic information relating to aircraft altitude, airspeed, and the surrounding atmosphere. Computer inputs consist of total and indicated static pressure from the pitot static system, total temperature from a remote temperature sensor probe and indicated angle-of-attack from angle-of-attack

transducer. These basic inputs are corrected and integrated to provide mach number, indicated airspeed, true airspeed, incremented mach number, pressure altitude, pressure altitude rate of change, incremental pressure altitude, true temperature, and true angle-of-attack. The CADC is an analog-type computer that requires 115-volt AC and 28-volt DC electrical power.

Air data computer outputs are provided for the following aircraft systems:

ALTIMETER (BAROMETRIC)	- pressure altitude
MAXIMUM SAFE MACH ASSEMBLY	- pressure altitude, mach number, true air temperature
AUTOMATIC FLIGHT CONTROL SYSTEM	- true angle-of-attack, incremental mach number, incremental pressure altitude
ENGINE MACH LEVER ACTUATOR	- mach number
SPIKE CAUTION LAMP	- mach number
INERTIAL NAVIGATION SYSTEM	- true airspeed, pressure altitude rate
AIRBORNE MISSILE CONTROL SYSTEM	- mach number, true angle-of-attack, pressure altitude, pressure altitude rate, total temperature
DATA LINK	- mach number, indicated airspeed, pressure altitude, true airspeed
VDIG	- pressure altitude rate, true airspeed, pressure altitude
IFF	- pressure altitude

CADC CONTROLS AND INDICATORS

CADC Power Switch

The CADC power switch on the ground check panel (figure 1-23) has two positions placarded POWER and OFF. When switch is in OFF, no aircraft power is supplied to CADC or maximum safe mach assembly, and CADC caution lamp on main caution lamp panel will illuminate. When switch is placed in POWER, 115-volt AC power is supplied to CADC and maximum safe mach assembly.

CADC Test Switch

The CADC test switch, with positions HIGH, OFF, and LOW, on ground check panel (figure 1-23) activates a self-test system. The normal system inputs are disconnected from CADC, and a set of pre-selected test inputs are fed into CADC. Normally this switch is used by flight crew only during functional or acceptance check flights.

CADC Caution Lamp

A computer monitor system provides an advisory lamp on main caution lamp panel (figure 1-28). The lamp illuminates CADS, indicating air data system power failure, servo loop malfunction, or failure of maximum safe mach assembly. Reliable pressure altitude signals from the computer provide monitor failure interlocks for barometric altimeter.

AUXILIARY FLIGHT REFERENCE SYSTEM (AFRS)

The auxiliary flight reference system (AFRS) provides standby or backup attitude and directional information. The system consists of a number of electronic packages which receive, compute, and transmit gyroscopic attitude and directional reference signals. Basic components of the system include a two-gyro three gimbal platform, a control amplifier, a compass controller, and a remote compass transmitter (flux valve). The platform is unlimited in roll but is limited to ±82 degrees in pitch. Any change in aircraft attitude with respect to vertical reference is detected by platform and electrically transmitted to attitude director indicator when the system is operating in the standby mode. The directional gyro in the platform and flux valve operate together as a compass set to provide heading signals to HSI and ADI when the system is operating in the standby mode. The compass set operates either as a gyro-stabilized magnetic compass (slaved mode), as a directional gyro (DG mode), or as a non-gyro-stabilized compass using the servo in the control amplifier (compass mode). The two modes, slaved and DG, provided accurate heading reference for all latitudes. In the slaved mode, the system is basically a directional gyro slaved to the remote compass transmitter. This mode is designed for use at latitudes up to 70 degrees. In the polar regions, direction of the earth's magnetic field becomes more vertical rather than horizontal to such an extent that the slaved mode is not reliable and the DG mode should be used. In the DG mode, the system is freed from the remote compass transmitter and operates as a free gyro, indicating an arbitrary gyro heading. In the DG mode, apparent drift due to earth's rotation is corrected. The random drift (precession rate) of the gyro in the DG mode will not exceed ±1.5 degrees per hour. This mode may be used at all latitudes but is more useful when operating in the polar regions or when the magnetic field is weak or distorted. The compass (COMP) mode provides unstablized compass heading. The purpose of this mode is to permit continued operation of the AFRS if a malfunction of the gyros occurs. The AFRS operates on 115-volt AC power from the AC essential bus and 28-volt DC power from the DC essential bus.

AUXILIARY FLIGHT REFERENCE SYSTEM POWER SWITCH

The auxiliary flight reference system power switch on ground check panel (figure 1-23) has two positions - GYROS and OFF. Placing switch to GYROS supplies power to the AFRS gyros and the compass set. Placing switch to OFF de-energizes these components.

REFERENCE SELECT PANEL

The reference select panel on forward portion of left console (figure 1-5) contains attitude heading switch (ATT/HDG) and gyros fast erect button (STBY GYROS FAST ERECT).

Attitude Heading Switch (ATT/HDG)

The attitude heading switch (ATT/HDG) on reference select panel has two positions placarded PRI and STBY. When in PRI (primary) position, pitch, roll and heading information from inertial navigation system (INS) is supplied to the following subsystems: Autopilot (pitch, roll, and heading); Horizontal Situation Indicator (heading); VDIG (pitch, roll, and heading).

Placing the switch to STBY (standby) provides information to all of the above subsystems from the auxiliary flight reference system (AFRS) except autopilot. The AFRS supplies pitch and roll information to the attitude Direction Indicator.

Gyros Fast Erect Button (STBY GYROS FAST ERECT)

The gyros fast erect button on reference select panel provides a means for fast erection of the AFRS gyros. The button is placarded STBY GYROS FAST ERECT. During initial turn-on, gyros will automatically erect at the fast rate of approximately 15 degrees per minute. If re-erection is required after initial turn-on, due to limits of gyro being exceeded, fast erection may be accomplished by depressing and holding STBY GYROS FAST ERECT button until gyros return to normal. During initial erection or when STBY GYROS FAST ERECT button is depressed, STANDBY ATTITUDE caution lamp on main caution lamp panel will illuminate; and, if attitude heading switch is in STBY, OFF flag on attitude direction indicator (ADI) will come into view. Normally gyros erect at a rate of approximately 5 degrees per minute.

PRIMARY ATTITUDE/HEADING CAUTION LAMP

The primary attitude/heading caution lamp on main caution lamp panel (figure 1-28) is placarded PRI ATT/HDG. The lamp will illuminate when INS is unreliable or attitude heading switch is in STBY.

STANDBY ATTITUDE CAUTION LAMP

The standby attitude caution lamp on main caution lamp panel (figure 1-28) is placarded STANDBY ATTITUDE. The lamp will illuminate when attitude information from AFRS becomes unreliable. The standby attitude lamp will also illuminate during initial gyro erection, and when STBY GYROS FAST ERECT button is depressed.

COMPASS PANEL

The compass control panel (figure 1-27) on left console provides necessary controls for AFRS heading reference.

Compass Mode Selector

The compass mode selector on compass panel (figure 1-27) is used to select mode of operation of auxiliary flight reference system compass. The selector has three positions marked SLAVED, COMP, and DG. When the SLAVED mode is selected, gyro stabilized magnetic heading from the remote compass transmitter is provided. In the DG mode, remote compass transmitter information is removed from the system and operates as a free gyro, indicating an arbitrary gyro heading. In the COMP mode, compass heading is obtained directly from remote compass transmitter without stabilization by the directional gyro, and is used if an attitude malfunction of the auxiliary flight reference system occurs.

Note

> When moving the selector from the SLAVED position to COMP, compass card on HSI will rotate off heading and immediately return. This is normal. When moving selector from COMP back to SLAVED, compass card of HSI will rotate off heading but will not return until heading set selector is depressed and held to null synchronization indicator.

Latitude Correction Selector

The latitude correction selector on compass panel (figure 1-27) is marked with latitudes from 0 degrees to 90 degrees. Setting selector to latitude at which

COMPASS PANEL

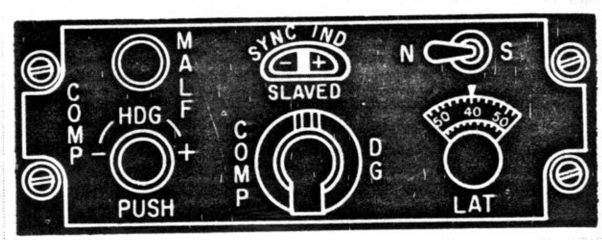

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Figure 1-27

flight is being made determines rate of gyro drift correction when operating in DG mode and improves accuracy when operating in SLAVED mode.

Heading Set Selector

The heading set selector on compass panel (figure 1-27) provides a means of rapidly synchronizing AFRS gyro with remote compass transmitter when operating in SLAVED mode, and to set in desired heading on HSI when operating in DG mode. When compass is operated in SLAVED mode, fast synchronization is accomplished by depressing and holding selector depressed until synchronization indicator on compass control panel becomes centered. When compass is operated in DG mode, system heading is changed by depressing and turning selector to right to increase heading and left to decrease heading. The rate of heading change is determined by the amount selector is turned. When compass is operated in COMP mode, system continuously tracks remote compass transmitter, and it is not necessary to use selector.

Hemisphere Selector Switch

The hemisphere selector switch on compass panel (figure 1-27) has two positions marked N (North) and S (South). The switch must be positioned to correct hemisphere in which aircraft is operating to provide proper polarity of earth's rate correction.

Synchronization Indicator

The synchronization indicator on compass panel (figure 1-27) indicates whether or not AFRS gyro and remote compass are synchronized. During operation in SLAVED mode, pointer will normally fluctuate slightly when compass set is synchronized with gyro. Should compass get out of synchronization, pointer will deflect toward either plus or minus sign on face of indicator. To synchronize system, heading set selector must be depressed and held until pointer is centered to synchronize the system. The indicator is de-activated when operating in DG or COMP modes.

Heading Malfunction Caution Lamp

An amber heading malfunction caution lamp on compass panel (figure 1-27) is provided to indicate that AFRS heading is unreliable. A push-to-test circuit is provided to check lamp.

PITOT-STATIC SYSTEM

The pitot-static system transmits impact (pitot) air pressure and atmospheric (static) pressure to central air data computer, airspeed/mach number indicator, vertical velocity indicator, and servo barometric altimeter. Two probes, one on forward left side and one on forward right side of aircraft are pick-up points for pitot-static pressure. The left probe picks up only static pressure and right probe picks up both static and pitot pressures. Both probes are equipped with heating elements to prevent icing. The heating elements are controlled by probe heaters switch on external environment panel (figure 1-34) on left console.

WARNING, CAUTION AND INDICATOR LAMPS

The warning, caution and indicator lamps are shown in figure 1-28. Illumination of a lamp is applicable to a particular system and a condition associated with it. The description and function of each light is incorporated under the applicable system.

MASTER CAUTION LAMP PANEL

The master caution lamp panel (figure 1-28) on lower center portion of pilot's instrument panel provides caution lamps for various aircraft systems. Illumination of a particular lamp is advisory of a malfunction or emphasizes a particular condition of the applicable system, and will cause master caution lamp to flash. When master caution lamp is reset (PRESS TO RESET), caution lamp will remain on until discrepancy is corrected. When reset, master caution lamp is rearmed for any subsequent caution lamps.

MASTER CAUTION LAMP

The master caution lamp on the upper right side of pilot's instrument panel (figure 1-3) is an amber lamp that displays MASTER CAUTION PRESS TO RESET when illuminated. The master caution lamp flashes when one of the lamps on master caution lamp panel is illuminated. When this master lamp is reset (PRESS TO RESET), it goes out and is rearmed for illumination of other caution lamps; but appropriate caution lamp on master caution lamp panel remains illuminated until discrepancy is corrected.

CREW MODULE

The crew module forms an integral portion of the forward fuselage, encompassing a pressurized crew compartment and forward portion of wing glove (figure 1-29). It is ejected from aircraft as a complete unit, providing a safe escape system for crewmembers during emergencies. It is also capable of underwater ejection, and provides maximum protection and survival from environmental hazards on either land or water. An emergency oxygen supply system and a self-contained emergency pressurization system are provided primarily for use during ejection. However, either system can be manually activated in flight as an auxiliary to aircraft primary system.

Note

> The addition or deletion of components in the crew module or the absence of a crewmember will change CG of module and adversely affect zero altitude ejection capability.

CANOPY

The canopy consists of left and right clam shell hatches hinged to a center beam assembly. The hatches open to a maximum of 65 degrees. Each hatch has an external and internal canopy latch handle for opening or closing. When the hatches are closed and latched, internal handle locks in place to prevent

WARNING, CAUTION AND INDICATOR LAMPS

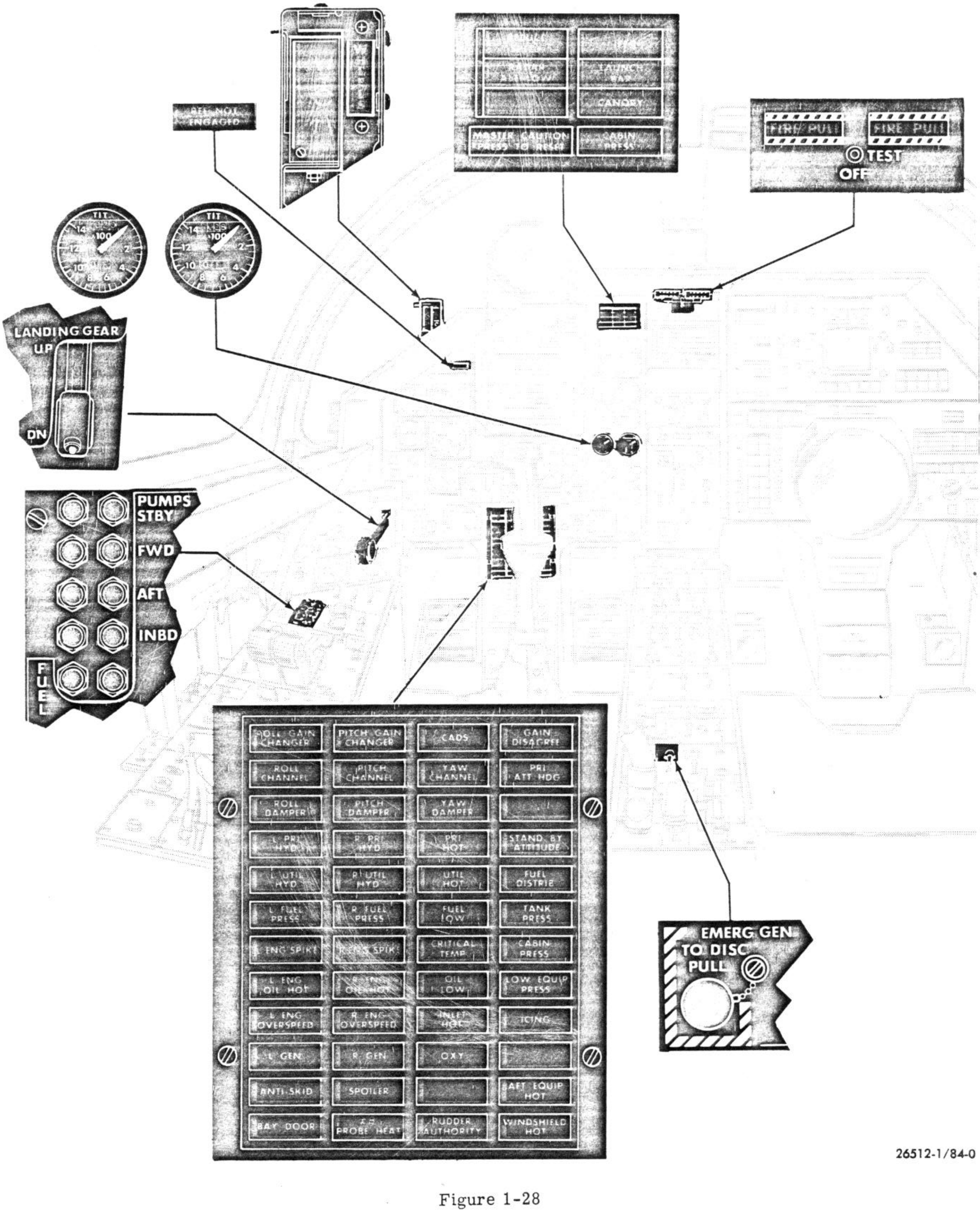
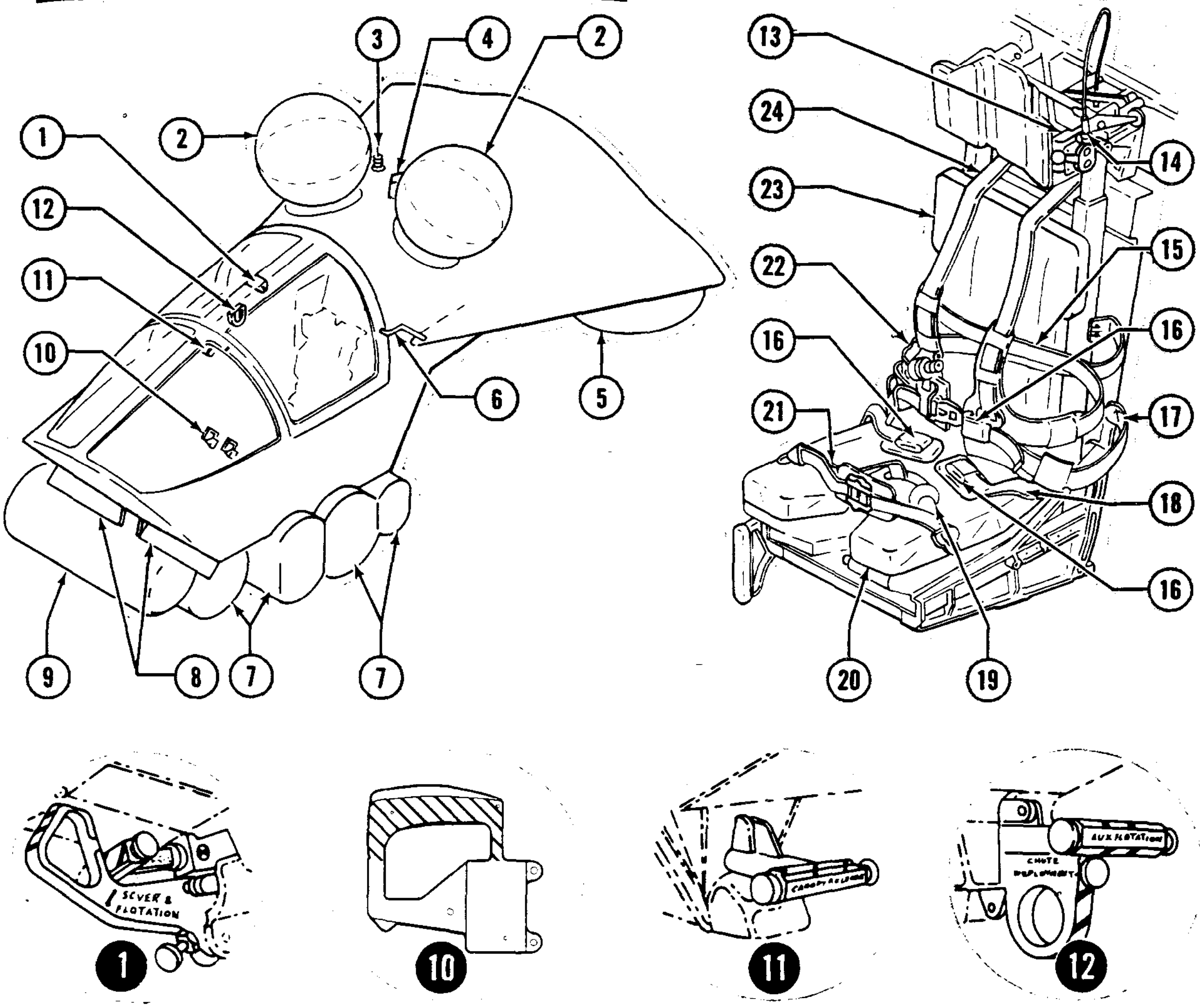


Figure 1-28

CREW MODULE AND SEAT



1. PARACHUTE RELEASE AND SEVERANCE AND FLOTATION HANDLES
2. SELF-RIGHTING BAGS
3. EMERGENCY UHF ANTENNA
4. UHF ANTENNA
5. AFT FLOTATION BAGS
6. EMERGENCY VENTILATION TUBE
7. IMPACT ATTENUATION BAG
8. CHIN FLAPS
9. FWD AUXILIARY FLOTATION BAG
10. CREW MODULE EJECTION HANDLES
11. CANOPY INTERNAL RELEASE HANDLE
12. PARACHUTE DEPLOY AND AUXILIARY FLOTATION HANDLES
13. HEADREST ADJUSTMENT LEVER (2)
14. INERTIA REEL CONTROL HANDLE
15. CHEST STRAP
16. QUICK RELEASE BUCKLE (3)
17. TRUNK STRAP
18. LAP BELT
19. CROTCH STRAPS
20. SEAT FORE AND AFT ADJUSTMENT LEVER
21. LOWER TORSO RESTRAINT HARNESS
22. UPPER TORSO RESTRAINT HARNESS
23. SEAT BACK CUSHION
24. SHOULDER STRAP

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Figure 1-29

inadvertent unlatching of hatch in flight. Each hatch is manually raised or lowered with aid of an air/oil counterpoise. The counterpoise will also hold hatch in any position selected. The left canopy hatch can be explosively detached along canopy center beam to allow crew rescue from above by helicopter.

Canopy Internal Latch Handles

Two canopy internal latch handles are on inside lower horizontal frame member of each canopy hatch. An over-center, spring-loaded latch handle lock tab, in face of each latch handle, locks handle in latched position to prevent inadvertent opening in flight. The canopy hatch handle is locked when this tab is flush with the surface of handle. Pressing in on forward part of lock tab will cause rear part of tab to snap out, unlocking latch handle. The handle must be pulled out and aft to a detent position to unlatch the hatch. Once the hatch is unlatched, pulling handle further aft past detent engages counterpoise to aid in opening. Releasing handle when desired hatch opening is reached will allow it to return to detent position and lock counterpoise. This will hold hatch at the opening selected. When hatch is closed, canopy pressurization seals are automatically inflated and canopy unlock warning lamp is turned off.

Canopy External Latch Handles

Two flush-mounted canopy external latch handles are on lower horizontal frame member of each canopy hatch. Each handle is mechanically linked to its respective internal handle. Pressing in on forward part of handle will extend rear portion of handle so that it may be grasped to unlatch and raise the hatch. If internal handle is locked in closed position, hatch cannot be opened from outside.

Left Canopy Detach Handle

The left canopy hatch can be detached along center canopy beam by pulling left canopy detach handle on aft bulkhead above pilot's seat. Pulling the handle fires an initiator, which in turn, fires an explosive charge to separate left canopy hatch along center beam. The handle is marked CANOPY DETACH. A safety pin is inserted in handle to prevent inadvertent actuation.

Canopy Unlock Warning Lamp

A red canopy unlock warning lamp on pilot's instrument panel (figure 1-3) will illuminate when either hatch is not locked. When illuminated, word CANOPY is visible on face of lamp.

SEATS

The crew module seats (figure 1-29) are electrically adjustable vertically, and manually adjustable forward and aft. The headrest, attached to aft bulkhead, and seat pan are manually adjustable forward and aft. Forward adjustment of headrest requires inertia reel to be unlocked. The seat back is attached to back of seat pan by pivot pins and attached to headrest by pivot pins. Therefore, any forward or aft adjustment of seat pan and headrest will cause seat back to tilt.

Seat Adjustment Switches

Vertical adjustment of each seat is controlled by a three-positioned, momentary, contact switch on sidewall adjacent to each seat. Each switch is marked UP and DOWN and spring-loaded to the center position (OFF). The direction of switch movement corresponds to direction of seat movement. Maximum vertical travel of seat is 5 inches.

Note

The seat actuator motor is an intermittent type with a 1 minute on and a 19 minute off cycle.

Seat Forward And Aft Adjustment Lever

The seat forward and aft adjustment lever is in front of seat pan. When handle is pulled up, seat is unlocked from carriage and allows a maximum travel of 5 inches from full aft to full forward. Forward or aft adjustment will result in a tilting of the seat back.

Seat Headrest Adjustment Lever

A seat headrest adjustment lever (figure 1-29) on either side of each seat headrest is provided for fore and aft adjustment of headrest. Depressing either lever unlocks headrest and allows it to be moved forward or aft. Releasing lever will lock headrest in place. Since seat back is attached to headrest, fore and aft movement of headrest will cause seat back to tilt.

Seat Harness

Each seat is equipped with an upper and a lower torso harness. The upper harness consists of shoulder straps, adjustable chest straps, and trunk straps. These connect at the center of crewmember's chest by a quick release buckle. The chest and trunk straps are attached to seat and shoulder straps attached to inertia reel. The lower torso harness consists of an adjustable lap belt attached to each side of seat pan and is connected by a quick release buckle.

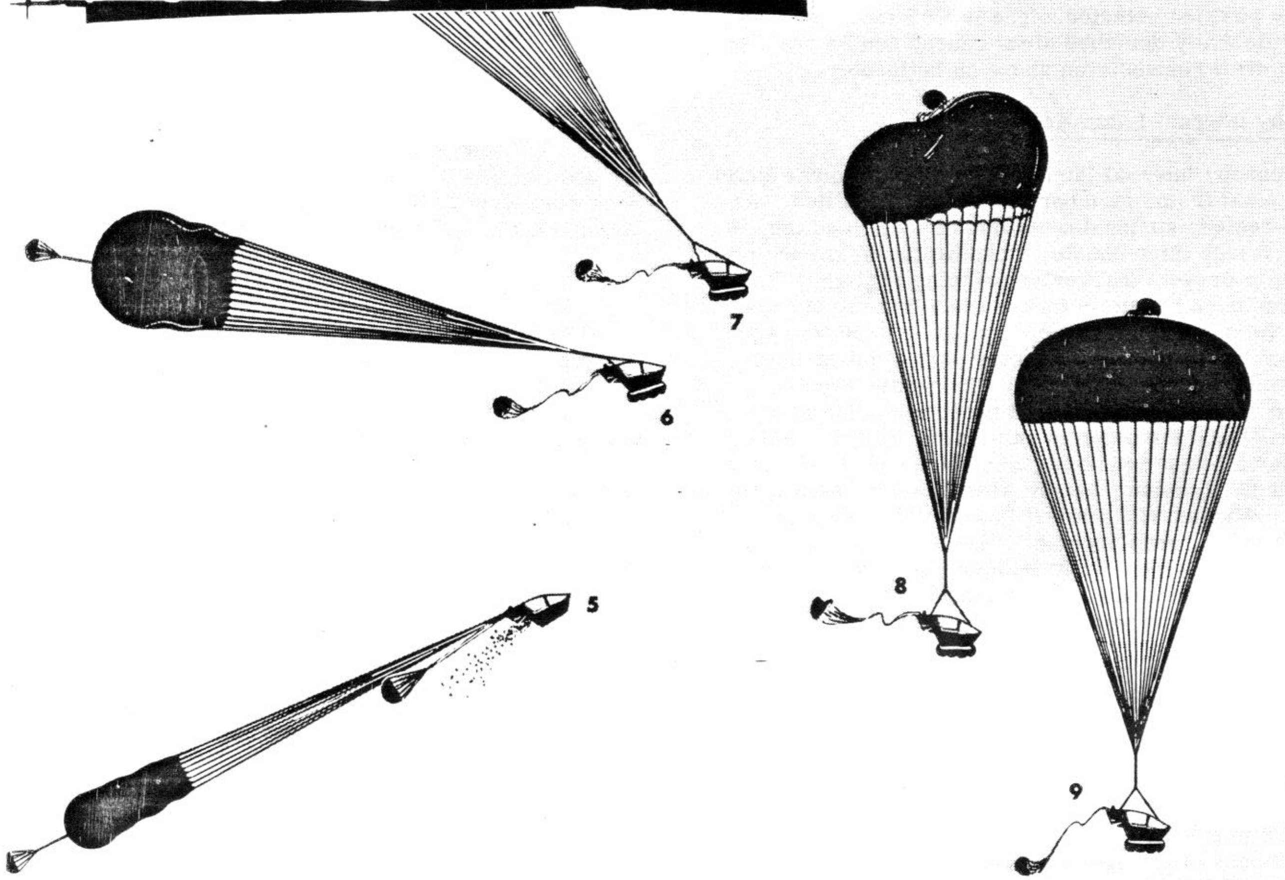
Inertia Reel

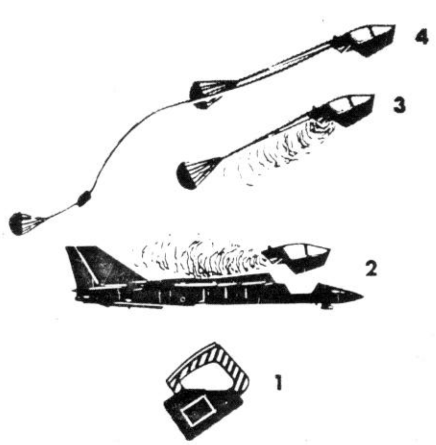
Each seat provides an inertia reel behind headrest. The inertia reel control handle on left side of each seat headrest locks or unlocks inertia reel. When unlocked, inertia reel allows should straps to extend or retract and provides crewmember freedom of movement. When locked, either manually or by a g-force, inertia reel prevents shoulder strap extension and takes up slack as crewmember returns to a normal position. During module ejection, inertial reel is locked and shoulder straps retracted by automatic activation of an explosive cartridge.

EJECTION SEQUENCE

Actuation of either ejection handle initiates explosive ejection sequence (figure 1-30); simultaneously

CREW MODULE EJECTION SEQUENCE





1. INITIATE EJECTION	TIME: 0.000 SEC
INERTIA REEL LOCKS, OXYGEN AND PRESSURIZATION ACTUATES	0.005 SEC
2. CREW MODULE SEVERS	
ROCKET MOTOR FIRES	0.35 SEC
MODULE ENTERS SLIPSTREAM	0.48 SEC
STABILIZATION—BRAKE CHUTE DEPLOYS	0.50 SEC
ROCKET MOTOR SECONDARY NOZZLE SEVERS	0.50 SEC (300-800kts)
3. ROCKET MOTOR BURNS OUT	1.15 SEC (0-300kts) 1.45 SEC (300-800kts)
4. RECOVERY CHUTE DEPLOYS	1.35 SEC (0-300kts) 1.6 to 4.4 SEC (300-800)
5. CHAFF DISPENSES	3.00 SEC
RECOVERY CHUTE LINE STRETCH	0.7 to 3.0 SEC AFTER RECOVERY CHUTE DEPLOYS
6. IMPACT ATTENUATION BAG DEPLOYS	3.0 SEC AFTER RECOVERY CHUTE DEPLOYS
7. RECOVERY CHUTE DISREEFS	2.5 SEC AFTER RECOVERY CHUTE LINE STRETCH
8. CREW MODULE REPOSITIONS IMPACT BAG INFLATED	7.0 SEC AFTER RECOVERY CHUTE DEPLOYS
9. RECOVERY CHUTE FULL BLOSSOM	11.0 TO 15.0 SEC

NOTE

THIS EJECTION TIME SEQUENCE IS BASED ON AN AIRCRAFT AIRSPEED OF 0 TO 300 KTS EXCEPT AS OTHERWISE NOTED

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Figure 1-30

retracts and locks shoulder harness inertia reels, activates emergency oxygen and crew module pressurization systems, activates chaff dispenser (if armed), fires explosive guillotine cutters, and ignites rocket motor. Pressure build-up of rocket motor fires two additional initiators, first of which initiates detonation of flexible linear-shaped charges that separates the module from aircraft. It also acts as a back-up system to guarantee activation of the emergency oxygen and crew module pressurization systems, the activation of chaff dispenser, and detonation of explosive guillotines. The explosive guillotine cutters sever antenna leads, secondary control cables and oxygen lines. Quick disconnects in module floor separate the air conditioning and pressurization ducts, flight controls, and electrical wiring. The rocket motor between crew stations and behind seat bulkhead propels the module up and away from aircraft. The second initiator actuates stabilization-brake parachute, thrust reducer, and unlocks barostat initiator. The stabilization glove (forward portion of the wing glove) serves to stabilize crew module during ejection. Pitch flaps, in the under surface of glove section and chin flaps along bottom of forward pressure bulkhead, assist in maintaining horizontal stability. A stabilization brake parachute is used to decelerate module and assists in maintaining stable flight until the recovery parachute is deployed. The height attained by module at its apogee will vary with the airspeed and altitude at time of ejection. During level flight, this height may vary from 400 feet at zero knots to more than 900 feet at 800 knots.

Note

Ejection above 700 knots is not recommended.

When the barostat initiator is unlocked and senses an altitude below 15,000 feet, the recovery parachute cover is removed and the parachute catapult fired. This deploys recovery parachute in a reefed configuration at a timed interval, and after line stretch is reached, the reefing line is severed and parachute disreefs. The barostat initiator also removes the impact attentuation bag cover and fires the pneumatic air bottles, causing attentuation bag to inflate. A third function of the barostat initiator is to erect emergency UHF antenna and fire explosive pin retractor, releasing repositioning bridle cable so the module will assume correct touchdown attitude. Ground or water landing impact is absorbed by controlled gas expulsion from impact landing bag. After landing, recovery parachute is released from module by pulling recovery parachute release handle. The severance and flotation handle guards release handle and must be pulled before recovery parachute can be released.

Ejection Handles

There are two ejection handles, one on each side of center console, adjacent to each crewmember's seat. Depressing lock release on top of handle allows handle to be pulled out, which, when pulled an approximate 3/4-inch distance, initiates the ejection sequence.

Parachute Deploy Handle

The ring-shaped recovery parachute deploy handle on canopy center beam assembly is used as an emergency means of deploying the recovery parachute, should the normal method fail. Pulling handle will fire both recovery parachute and stabilization brake parachute catapults.

Severance And Flotation Handle

The severance and flotation handle, placarded SEVER & FLOTATION, is on canopy center beam. It provides manual activation of module severance system to separate the crew module from aircraft after ditching. Pulling the handle also activates emergency oxygen system, and inflates the aft flotation and self-righting bags. A release button positioned on bottom of handle must be depressed to unlock handle.

Parachute Release Handle

The ring-shaped recovery parachute release handle on canopy center beam assembly is provided to release the recovery parachute from crew module after landing. Pulling handle fires parachute release retractors at bridle attaching points, releasing bridles from crew module. The recovery parachute release handle cannot be pulled until severance and flotation handle has been pulled.

Auxiliary Flotation Handle

The T-shaped auxiliary flotation handle on canopy center beam assembly is provided to inflate the auxiliary flotation bag. Pressing a release button on either side of handle and pulling handle out fires an initiator which, in turn, removes the severable cover over the auxiliary flotation bag and fires an explosive valve in an air storage bottle to inflate bag.

Bilge/Flotation Bag Inflation Pump

The bilge/flotation bag inflation pump is used to simultaneously pump water from the crew module and to maintain inflation of flotation bags. Overinflation of bags is prevented by relief valves. After landing, the bilge/flotation pump drive connector pin is removed from pin stowage hole and inserted in the operating hole. This connects pump to control stick. A plunger, adjacent to pin stowage hole, must be pushed in to open pump air and water outlet valves. Fore and aft motion of control stick will then operate pump.

Flotation and Landing Equipment

Even though the crew module is water tight and will float, additional bouyancy is provided by flotation and self-righting bags. An impact attenuator bag under crew module floor serves to cushion landing impact. The flotation bags consist of an auxiliary bag on front of the module and two aft bags (one attached to each aft corner of glove section). The self-righting bags consist of two bags, one on top of each side of glove section; when inflated in sequence, they assist in uprighting a possibly overturned module and keeping

it upright in water. Pneumatic storage bottles supply the necessary pressure for inflating all of the bags. The auxiliary flotation bag is inflated by pulling a T-shaped AUX FLOTATION handle. The aft flotation bags and the self-righting bags are inflated by pulling the SEVER FLOTATION HANDLE or are automatically inflated by the underwater severance initiator system. The impact attenuation bag is automatically inflated during descent by the module ejection sequence system.

MODULE DITCHING AND SEA RECOVERY

If aircraft is ditched, automatic safeguards provide crew module severance, inflation of the aft flotation and self-righting bags, and actuation of the emergency oxygen system. The underwater severance initiator is actuated when submerged to a depth of 10 to 20 feet. Manual actuation is accomplished by pulling the severance and flotation handle on canopy center beam. Deployment of self-righting bags assures an upright position if the module surfaces inverted or capsizes after a water landing. Aft flotation bags provide sufficient free-board at sea to allow opening the canopies for ventilation. The auxiliary flotation bag provides additional buoyancy and free-board. If high seas prevent canopy opening, a snorkel ventilation system allows crew to breathe outside air. When deployed, flotation bags provide sufficient buoyance to support a swamped module. After a water landing, the control stick can be converted to manually actuate a combination bilge/flotation bag.

EMERGENCY PRESSURIZATION SYSTEM

The crew module escape system incorporates an emergency pressurization system. The system operates automatically during ejection to maintain pressurization of the module and canopy hatch seals. Should the automatic feature fail, system is manually activated with an emergency pressurization handle. Also, this system can be used as an alternate pneumatic supply source for pressurization of the crew module and canopy hatch seals if failure of normal pressurization system occurs. Pressure for the system is contained in a 650-cubic inch storage bottle behind the seat bulkhead. When activated, an aneroid-operated absolute pressure regulator, which senses cabin altitude, will open if cabin altitude is above 24,000 feet. Volume of storage bottle is sufficient to maintain this cabin altitude for approximately 4 minutes at maximum ejection altitude.

Emergency Pressurization Handle

The emergency pressurization handle on upper right corner of aft bulkhead console (figure 1-8) is used to manually activate emergency pressurization system. Pulling handle out will open aneroid-operated absolute pressure regulator.

Emergency Pressurization Gage

An emergency pressurization gage on aft bulkhead console (figure 1-8) indicates pressure within emergency pressurization system storage bottle. The gage is calibrated from 0 to 4000 psi in 500 psi increments.

SURVIVAL EQUIPMENT

The survival equipment includes a chaff dispenser, an emergency radio, two air ventilation masks, two MK 2 life vests, and miscellaneous survival equipment. The chaff dispenser, when armed, will activate to dispense chaff automatically during ejection sequence.

Note

The chaff dispenser may be armed or disarmed prior to ejection as the situation requires.

Chaff Dispenser Control Lever

The chaff dispenser control lever on aft bulkhead (figure 1-8) is used to arm or disarm the crew module chaff dispenser. The lever is labeled CHAFF and has two positions marked ON and OFF. Placing lever to ON opens a mechanical interrupt to allow explosive train propagation to chaff release mechanism. When the crew module is ejected, the explosive train releases the chaff dispenser and the slip stream dispenses the chaff. Placing lever to OFF closes the mechanical interrupt, thereby disarming dispenser.

Emergency Radio

The AN/URT-27 radio in right console (figure 1-7) provides an intermittent modulated tone to assist in rescue operations. The radio may be operated with its own retractable antenna or connected to the crew module emergency UHF antenna.

Air Ventilation Masks

Two air ventilation masks, with life vests in a separate survival equipment stowage compartment adjacent to right console, are used when canopy hatches must remain closed because of rough seas or inclement weather. The mask hoses may be connected to air mask connector valves adjacent to crew seats. An air supply tube leads from each connector valve to an outside opening well above the water line.

Miscellaneous Survival Equipment (Flight Test Installation)

Miscellaneous survival equipment is stored in the survival equipment stowage compartment behind right seat. Access to compartment is gained by removing right seat headrest from its tracks, rotating seat forward and disconnecting seat actuator. The following equipment is contained in the area indicated:

Nomenclature	Right Console	Survival Compartment
Bags, Water Stowage		2
Compass	2	
Mirror, Signal	2	
Dye Marker	2	6
Desalting Kit		2
Mask, Ventilation	2	

Nomenclature	Right Console	Survival Compartment
Radio Beacon		1
Heat Tabs W/Stove		2
Candle, long-burning		2
Distress Light (Strobe)		1
Pliers W/Side Cutter		1
Sun Hat		2
Lipstick Anti-chap		2
Distress Light (Life Vest)	2	
Knife, Pocket		2
Poncho		2
Life Preserver, MK-II	2	
Distress Signal MK-13		16
Sponge, Bailing		2
Rations PSK-2 Part I&II		2
Whistle	2	
Nylon Cord (100')		2
Shark Chaser	2	
Drinking Water (Can)		8
Salt Tablets (Box)		2
Sunburn Ointment		2
Anti-exposure Suit		2

ENVIRONMENTAL CONTROL SYSTEM

The environmental control system (figure 1-31) is composed of a pressurization system and an air-conditioning system. These two systems combine to provide temperature-controlled and pressure-regulated air for heating, cooling, windshield defogging and pressurization for the crew module. Additionally, the system provides conditioned air for electronic equipment cooling, anti-g suits, pressure suits, anti-icing of engine inlets, windshield rain removal, windshield wash, vortex destroyer, wing and canopy seals. Hot air from the 16th stage of each engine is directed through an air to air heat exchanger, which is supplied with ram air through a scoop in the boundary layer bleed area. This heat exchanger provides the first stage of cooling for hot compressor bleed air. The cooled bleed air passes through a cold air modulating valve and an air to water heat exchanger to a cooling turbine. The energy required to drive this turbine is extracted from partially cooled bleed air, resulting in further reduction of temperature which provides system with a supply of pressurized cold air. The cold air leaving the cooling turbine passes through a water separator which removes approximately 80% of the free moisture.

Cabin temperature is selected by a crew-operated temperature selector control on the air-conditioning panel, and controlled by the temperature controller, temperature sensors, cold air modulating valve and a hot air modulating and shutoff valve. The hot air modulating and shutoff valve controls the mixing of partially cooled bleed air with cold air from the cooling turbine. The cold air modulating valve controls the volume of first stage cooled air admitted to the cooling turbine through the air to water heat exchanger. Temperature selected air then enters crew compartment through side diffusers and windshield defog outlets. A provision for connecting externally supplied air from a ground unit is on lower right side of fuselage aft of crew compartment. Emergency ram air can also be admitted to the crew module through a retractable ram air scoop on right side of fuselage.

PRESSURIZATION

Pressure in the crew module is controlled by a pressure-regulating valve in front of the crew module. Below 8,000 feet, pressure-regulating valve will remain full open, maintaining an unpressurized condition regardless of schedule selected (figure 1-32). Above 8,000 feet, pressure-regulating valve will modulate to maintain pressure to either the normal or combat schedule, as selected. A pressure safety valve at rear of cockpit module will relieve crew module pressure any time pressure exceeds 11 psi differential. If loss of normal system pressurization occurs, an emergency ram air scoop, which can be opened into the airstream, will admit ventilating air into the crew module and electronic equipment areas.

Pressurization Warning Lamp

A red pressurization warning lamp, marked CABIN PRESS, is on pilot's instrument panel (figure 1-3). The lamp will illuminate when cabin altitude is above 38,000 feet.

Pressurization Caution Lamp

An amber pressurization caution lamp marked CABIN PRESS is on main caution lamp panel (figure 1-28). The lamp will illuminate when cabin altitude is above 10,000 feet.

Cabin Altitude Indicator

A cabin altitude indicator on pilot's instrument panel (figure 1-3) displays cabin altitude.

ANTI-G SUIT

Each anti-g suit is connected to aircraft pressurization system by an anti-g suit hose on pilot's oxygen-suit control panel and the MCO's oxygen-suit control panel (figure 1-36). A test button marked anti-g test is for checking the operation of the anti-g suit valve. When button is depressed, anti-g suit bladders will inflate and when button is released, bladders will deflate.

ELECTRONIC EQUIPMENT COOLING

Environmental control system cooling air is ducted to the forward and aft electronic equipment bays and to individual electronic components as required. The cooling air is not temperature-regulated; however, the equipment bays are limited to 160°F (70°C). Overtemperature in the aft bay electronic equipment bay will illuminate an AFT EQUIP HOT caution lamp on the main caution lamp panel.

Aft Equipment Hot Caution Lamp

The amber caution lamp, marked AFT EQUIP HOT, is on main caution lamp panel (figure 1-28). The lamp

ENVIRONMENTAL SYSTEM

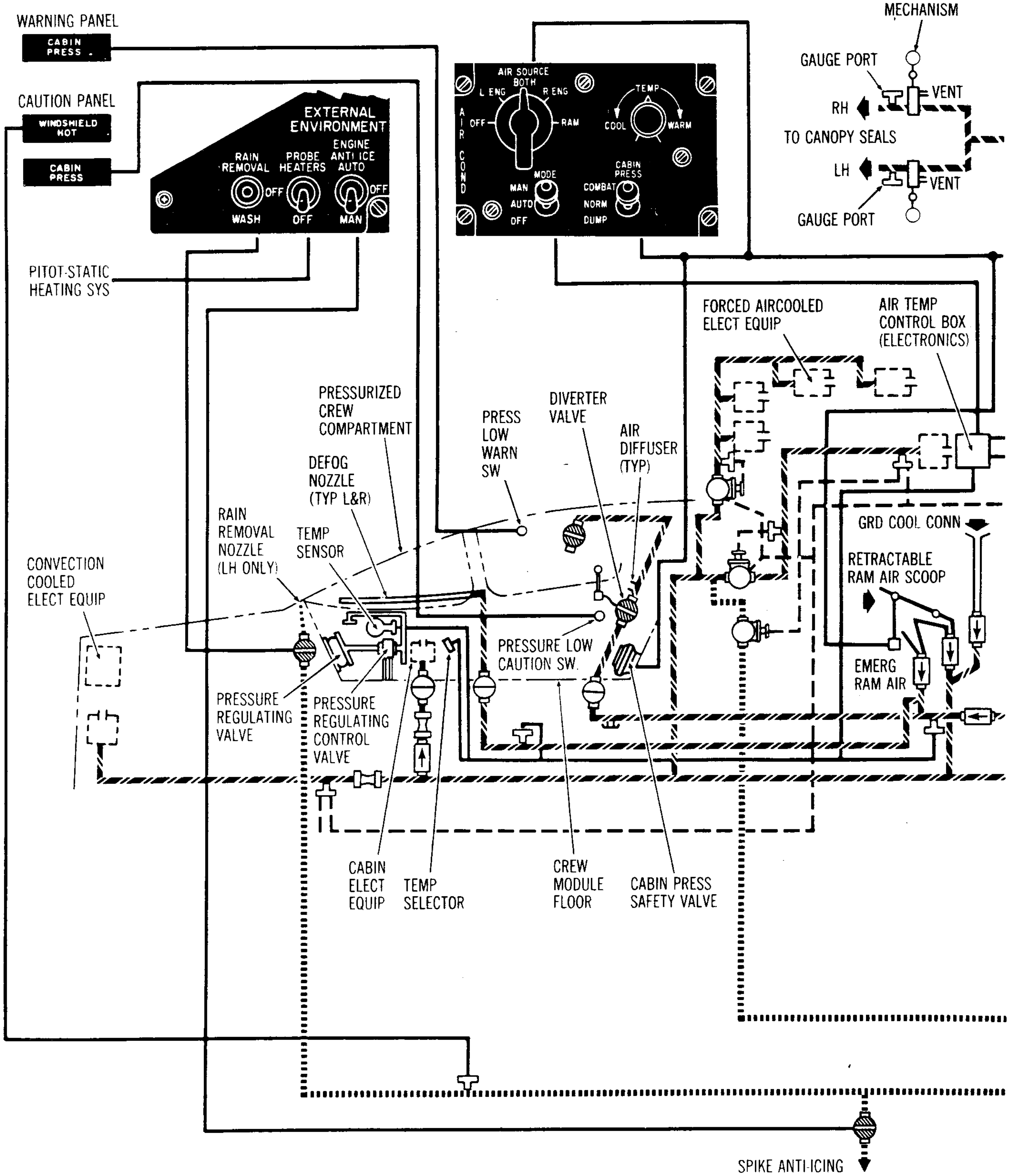

Figure 1-31 (Sheet 1)

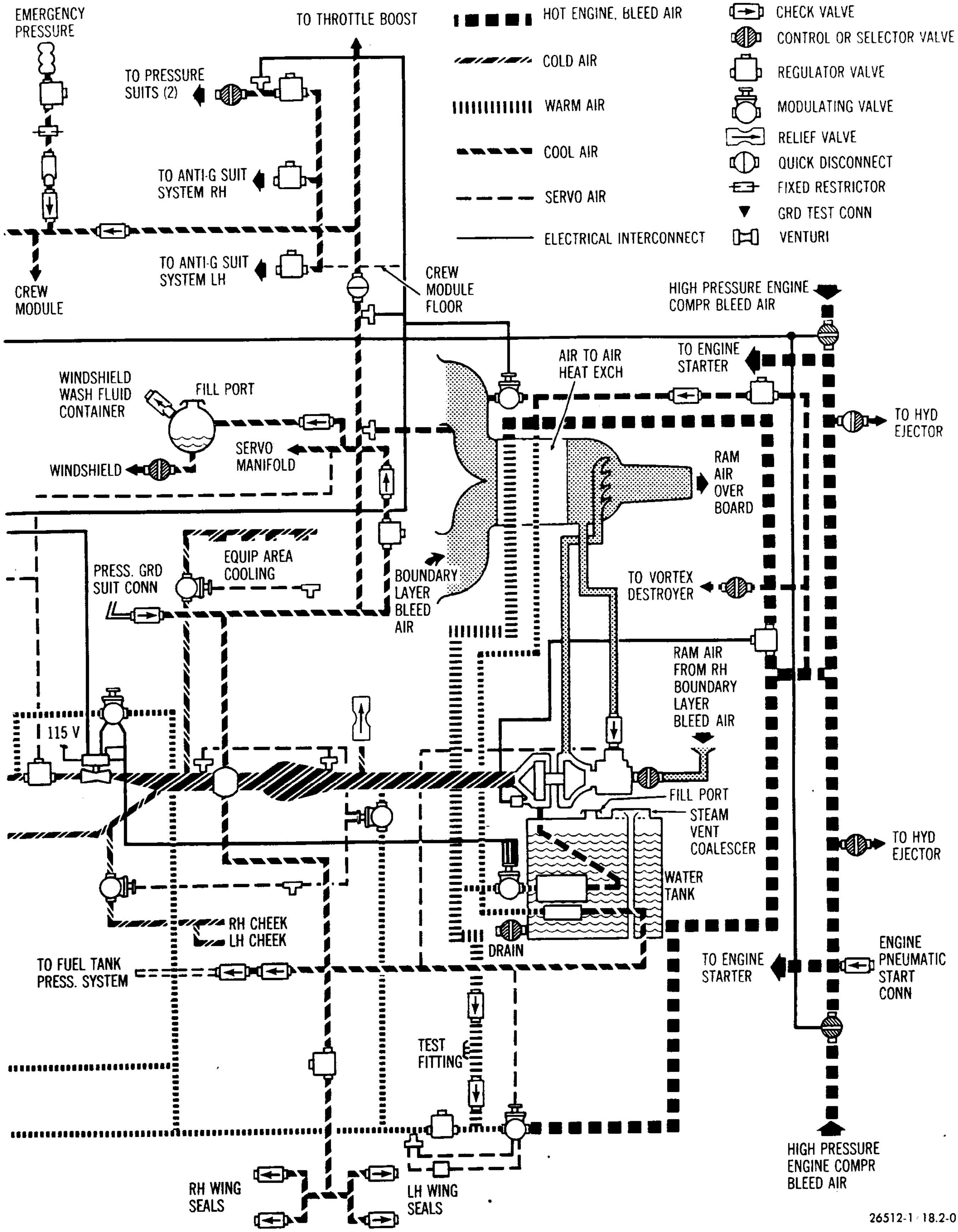

Figure 1-31 (Sheet 2)

CABIN PRESSURE SCHEDULE

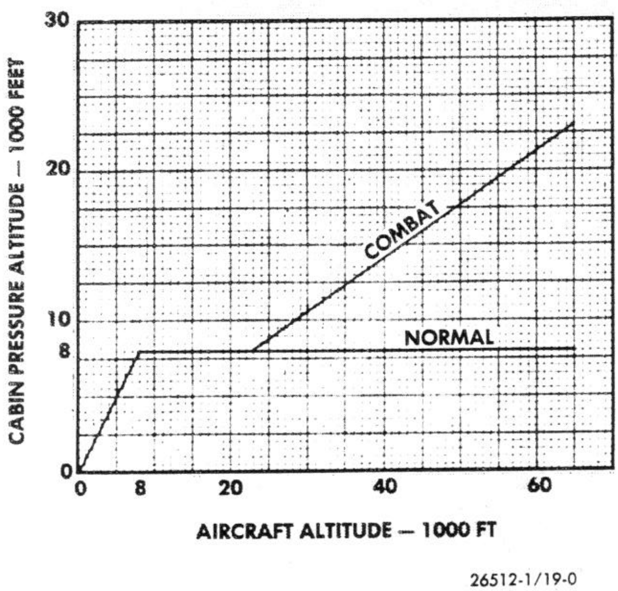

Figure 1-32

will illuminate if the cooling air flow is insufficient. When practicable, increasing engine rpm should supply sufficient air flow to correct the situation.

CAUTION

If the AFT EQUIP HOT lamp remains illuminated, turn off all non-essential equipment.

AIR CONDITIONING PANEL

The air conditioning panel (figure 1-33) is on aft section of center console and provides temperature and pressurization control switches.

Air Source Selector

The air source selector on air conditioning panel (figure 1-33) has five positions marked OFF, L ENG, BOTH, R ENG, and RAM. The knob controls bleed air source or allows selection of emergency ram air operation as required. In OFF, left and right engine bleed air shutoff valves are closed. In L ENG, left engine is the source of bleed air, and right engine bleed air shutoff valve is closed. In BOTH, left and right engine bleed air shutoff valves are open and supply bleed air to the environmental control system. In R ENG, right engine is the source of bleed air, and left engine bleed air shutoff valve is closed. In RAM, normal pressurization system pressure regulating valve is closed, ram air door is open, and crew module pressure regulating and relief valves are open. This will dump pressure and allow combined ram air flow and regulated engine bleed air to ventilate the crew module. Temperature control of this air is available by using temperature control selector to control amount of engine bleed air mixed with ram air. If air conditioning system fails to supply cooled air, ram air mode can be used for crew module and equipment cooling.

WARNING

To prevent excessive temperatures when pressure suits are worn, air conditioning system mode selector switch must not be placed to OFF prior to or while operating in RAM.

CAUTION

When operating in ram air mode, airspeed above 330 KIAS may result in structural failure of ram air door.

Air Conditioning Mode Selector Switch

The mode selector switch on air conditioning panel (figure 1-33) is a three-positioned switch marked AUTO, OFF, and MAN. In AUTO, crew module

AIR CONDITIONING PANEL

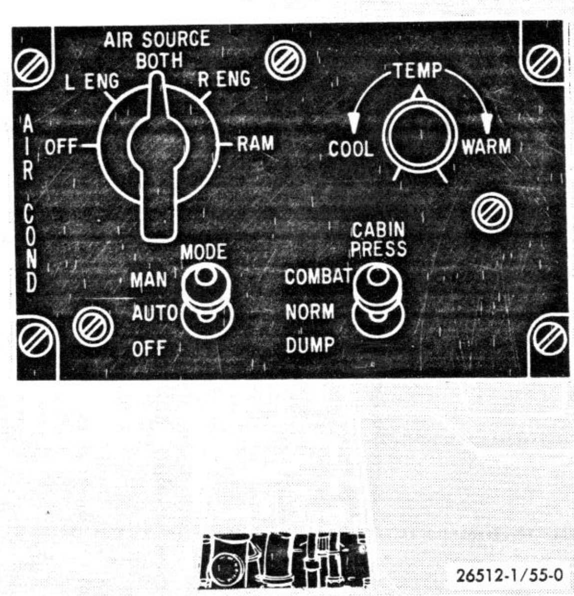

Figure 1-33

temperature is automatically controlled at temperature selected by temperature control selector. A signal goes to controller, which opens or closes modulating valves to maintain selected temperature. In MAN, temperature controller is by-passed and control of modulating temperature control valves is directly operated from temperature control selector. In OFF all power is removed from system, and valves in system which control temperature will declutch and go to full cool position. The valve controlling pressure suit ventilation temperature will remain in position it was in when power was removed.

Temperature Control Selector

The temperature control selector on air conditioning panel (figure 1-33) selects crew module temperature. The selector can be rotated through a 300 degree arc and has mechanical stops at each end. Extreme counterclockwise is marked COOL and extreme clockwise is marked WARM. With mode selector switch in AUTO, rotating selector in either direction sends a signal to temperature controller, which positions modulating temperature control valves to maintain selected temperature. The temperature is maintained at approximately 75° F (24° C) when temperature control selector is positioned at midpoint between COOL and WARM.

Note

> Operation with temperature control selector at full COOL in warm weather or full WARM in cool weather, with mode selector in AUTO, may result in an objectionable noise from the resulting high-speed air flow. The amount of airflow can be reduced by backing selector off full COOL or full WARM position.

With mode selector switch in MANUAL, signal goes directly to modulating temperature control valves, opening or closing them as directed by signal generated from temperature control selector. During manual operation, valves will respond only when selector is held against one of extreme positions, COOL or WARM. Maximum valve travel time from cool to warm is approximately 45 seconds.

Pressurization Selector Switch

The pressurization selector switch on air conditioning panel (figure 1-33) is a three-positioned lever lock switch labeled NORM, COMBAT, and DUMP. In NORM, crew module is selected to a schedule that will maintain an altitude of 8,000 feet up to operational ceiling of aircraft. In COMBAT, crew module pressure is maintained at an 8,000-foot level until aircraft reaches 23,000 feet, above which regulator maintains a 5 psi pressure differential. In DUMP, pressure safety valve and pressure-regulating valves are opened, resulting in an unpressurized condition in the crew module.

CABIN AIR DISTRIBUTION

Cabin air flow distribution is controlled by a lever on MCO's right console (figure 1-7) placarded CABIN AIR DISTR. It has two positions marked FWD DEFOG and AFT. The normal position of lever is in AFT. In this position, air flow into crew module is separated between air diffusers on rear bulkhead of crew module and windshield defog system, with approximately 85 percent directed to diffusers. Moving lever toward FWD DEFOG position will decrease airflow through air diffusers and increase airflow through defog system. When lever is in full forward position, all airflow will be directed through defog system. Although AFT position is considered normal to obtain maximum airflow, desired crew comfort is accomplished by selecting any intermediate position between FWD DEFOG and AFT.

WINDSHIELD DEFOG

Air for windshield defogging is controlled by cabin air distribution control lever on MCO's right console (figure 1-7). The lever labeled CABIN AIR DISTR has two positions marked FWD DEFOG and AFT. In FWD DEFOG, all airflow is directed through defog system. An intermediate position between FWD DEFOG and AFT may be selected as desired.

WINDSHIELD WASH AND RAIN REMOVAL SYSTEM

The windshield wash and rain removal system keeps left windshield clear of impinging rain and insects. Compressor bleed air at an approximate temperature of 390°F and at a pressure of 45 psi is directed over outside of windshield through a fixed area nozzle. This hot air blast will evaporate rain and prevent its further accumulation. Windshield wash is accomplished by injecting a liquid wash solution into rain removal nozzle which removes dirt and insects. The solution is contained in a one-half gallon tank on right side of nose wheel well. The tank is pressurized to 15 psi by regulated and cooled compressor bleed air.

Windshield Wash/Rain Removal Selector Switch

The windshield wash/rain removal selector switch on external environment panel (figure 1-34) has three positions marked RAIN REMOVAL, WASH, and OFF. The switch is spring-loaded from WASH to OFF, and is locked out of RAIN REMOVAL position. The switch must be pulled out to move from OFF to RAIN REMOVAL.

Placing switch to RAIN REMOVAL will open rain removal shutoff valve, allowing temperature-regulated and pressure-regulated compressor bleed air to be directed to left windshield. When switch is placed to WASH, a time delay relay is energized to open rain removal shutoff valve and windshield wash shutoff valve selected by windshield wash selector switch. While these valves are open, compressor bleed air and liquid windshield wash solution will be directed to left windshield. Positioning switch from WASH to OFF will close valves after a 5-second delay, shutting off air and windshield wash solution. When switch is in OFF, windshield wash and rain removal system is deenergized.

Windshield Hot Caution Lamp

The windshield hot caution lamp on main caution lamp panel (figure 1-28) indicates when windshield

temperature is above limits. An overheat switch, installed in rain removal air supply duct upstream of shutoff valve, will close whenever air temperature is above 450°F. When overheat switch closes, a circuit is completed to close rain removal shutoff valves and illuminate the windshield hot caution lamp. After switch closes, caution lamp will normally go out within 15 seconds.

ANTI-ICING SYSTEMS

Pitot-Static Probe Anti-Icing

The pitot static probes are equipped with heating elements powered by 115-volt AC essential bus. When energized, elements heat probes, thus preventing formation of ice.

Probe Heaters Switch

The PROBE HEATERS is a two-positioned switch marked OFF with an unmarked on and is on external environment panel (figure 1-34). When switch is placed to on, 115-volt AC power is furnished to heating elements in pitot-static tube, angle-of-attack and, when landing gear is retracted to total temperature probe.

To ground check the heater in total temperature probe, CADC test switch must be held to HIGH and probe heaters switch set to on. When switch is OFF, heaters are deenergized. The switch controls 28-volt DC power from main and essential buses.

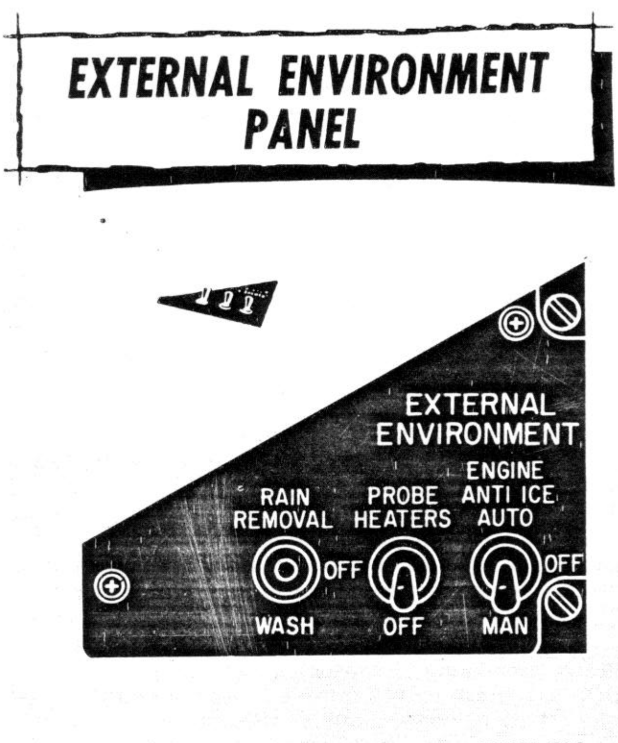

Figure 1-34

Engine Anti-Icing Switch

The three-positioned engine anti-icing switch on external environment panel (figure 1-34) is marked AUTO, MAN, and OFF. The lever lock-type switch locks in all three positions. In AUTO, anti-icing circuitry is armed, and when electronic ice detector senses an icing condition, a signal is transmitted to icing caution lamp. The signal also energizes a relay which turns on elements in spike sensing probe heaters and opens engine anti-icing and engine inlet anti-icing control valves, allowing circulation of hot air through anti-iced components. Approximately 60 seconds after icing condition ceases, hot air valves will close, spike probe heating elements will be deenergized and engine icing caution lamp will go out.

When switch is placed to MAN, engine anti-icing and engine inlet anti-icing valves open and spike probe heating elements are energized, whether or not ice detector senses an icing condition. Placing switch to OFF shuts off air to engine anti-icing and engine inlet anti-icing systems, and turns off spike probe heating elements; however, icing caution lamp will still be operational.

Engine Icing Caution Lamp

The engine icing caution lamp on main caution lamp panel (figure 1-28) will illuminate when electronic ice detector senses an icing condition. While icing condition exists, caution lamp will remain illuminated regardless of position of engine anti-icing switch. The lamp will go out 60 seconds after icing condition ceases.

Inlet Hot Caution Lamp

The inlet hot caution lamp on main caution lamp panel (figure 1-28) indicates that temperature of anti-icing bleed air to engine inlets has exceeded 450° (±10)F. When lamp illuminates, words INLET HOT are visible and anti-icing air to engine inlets is automatically shutoff, then lamp will go out.

OXYGEN SYSTEM

The oxygen system consists of a normal (liquid) system in forward fuselage and crew module, and an emergency (gaseous) system in crew module.

NORMAL OXYGEN SYSTEM

The normal oxygen system is designed for use with a pressure-demand oxygen regulator and mask to provide pressure-regulated 100 percent oxygen. See figure 1-35 for oxygen duration. Two converters, one on the left side of aircraft and one in nose wheel well, change liquid oxygen to gaseous oxygen, and regulates pressure from 70 to 80 psi during normal usage. From each converter, gaseous oxygen passes through a heat exchanger that warms it for breathing. The oxygen is then directed through a manually operated control valve a regulator and into face mask. Oxygen regulation is accomplished by a demand-type mini regulator, mounted on face mask. Evaporation loss will completely empty the fully serviced system in

approximately 10 days. The system has a twenty-liter capacity and is serviced through a single-point filler valve within an access door on left side of fuselage.

OXYGEN PANELS

There are two oxygen panels (figure 1-36), one on aft section of left console for pilot and one on aft bulkhead console for MCO. These panels contain oxygen control levers, oxygen hoses, anti-g suit hoses, anti-g suit test buttons, pressure suit hoses, suit vent knobs, and communication leads. The MCO's panel additionally contains emergency oxygen handle, emergency oxygen filler valve, and emergency oxygen gage.

Oxygen Control Levers

An oxygen control lever on each oxygen panel controls flow of oxygen from supply system to respective regulator. In ON, oxygen is supplied from converter to regulator. In OFF, oxygen flow is cut off at control valve on oxygen panel.

Oxygen Quantity Indicator

An oxygen quantity indicator, on pilot's instrument panel (figure 1-3) indicates total quantity of liquid oxygen in converter. The indicator dial is graduated from zero to 20 liters in increments of one liter. The indicator operates on 115-volt AC power from essential bus. If power should fail, indicator pointer will drive below zero, a fail safe indication.

Oxygen Quantity Indicator Test Button

A test button used for checking oxygen quantity indicator is just below fuel gage selector switch on pilot's instrument panel (figure 1-3) and is placarded OXY QTY TEST. When button is depressed, indicator pointer will move to zero, indicating system is operating properly. When button is released, pointer will move back to original reading. The oxygen caution lamp light will illuminate when pointer indicates a quantity of 2 liters or less.

Oxygen Caution Lamp

An amber caution lamp on main caution lamp panel (figure 1-28) will illuminate when oxygen quantity

OXYGEN DURATION

CABIN ALTITUDE	DURATION HOURS — LITERS OF LIQUID O_2										LESS THAN 1
	10	9	8	7	6	5	4	3	2	1	
35,000	31.2	28.1	25.0	21.8	18.7	15.6	12.5	9.4	6.2	3.1	EMERGENCY DESCEND BELOW 10,000' ACTUAL ALTITUDE
30,000	22.8	20.5	18.3	16.0	13.7	11.4	9.1	6.8	4.6	2.3	
28,000	20.4	18.4	16.3	14.3	12.2	10.2	8.2	6.1	4.1	2.0	
26,000	19.1	17.2	15.3	13.4	11.5	9.6	7.6	5.7	3.8	1.9	
24,000	16.5	14.9	13.2	11.6	9.9	8.3	6.6	5.0	3.3	1.6	
22,000	14.7	13.3	11.8	10.3	8.8	7.4	5.9	4.4	2.9	1.5	
20,000	13.3	12.0	10.6	9.3	8.0	6.6	5.3	4.0	2.7	1.3	
18,000	12.1	10.9	9.7	8.5	7.3	6.1	4.8	3.6	2.4	1.2	
16,000	11.1	10.0	8.9	7.8	6.7	5.6	4.4	3.3	2.2	1.1	
14,000	10.2	9.2	8.2	7.1	6.1	5.1	4.1	3.1	2.0	1.0	
12,000	9.3	8.4	7.5	6.5	5.6	4.7	3.7	2.8	1.9	0.9	
10,000	8.6	7.7	6.9	6.0	5.1	4.3	3.4	2.6	1.7	0.8	
8,000	7.8	7.0	6.2	5.4	4.7	3.9	3.1	2.3	1.5	0.7	
SEA LEVEL	5.5	4.9	4.4	3.8	3.3	2.7	2.2	1.6	1.1	0.5	

NOTE

- DURATION DATA SHOULD BE USED AS A GUIDE ONLY SINCE OXYGEN CONSUMPTION VARIES WITH THE INDIVIDUAL.
- CONVERSION OF LIQUID O_2 TO GASEOUS O_2 IS 860 LITERS OF GASEOUS TO 1 LITER OF LIQUID O_2.
- WHEN ONE PERSON IS USING OXYGEN, MULTIPLY THE NUMBER OF HOURS REMAINING BY TWO.
- CONSUMPTION RATES ARE BASED ON MIL I-9475A (USAF).
- CONSUMPTION RATES SHOWN ARE FOR 2 MEN.

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Figure 1-35

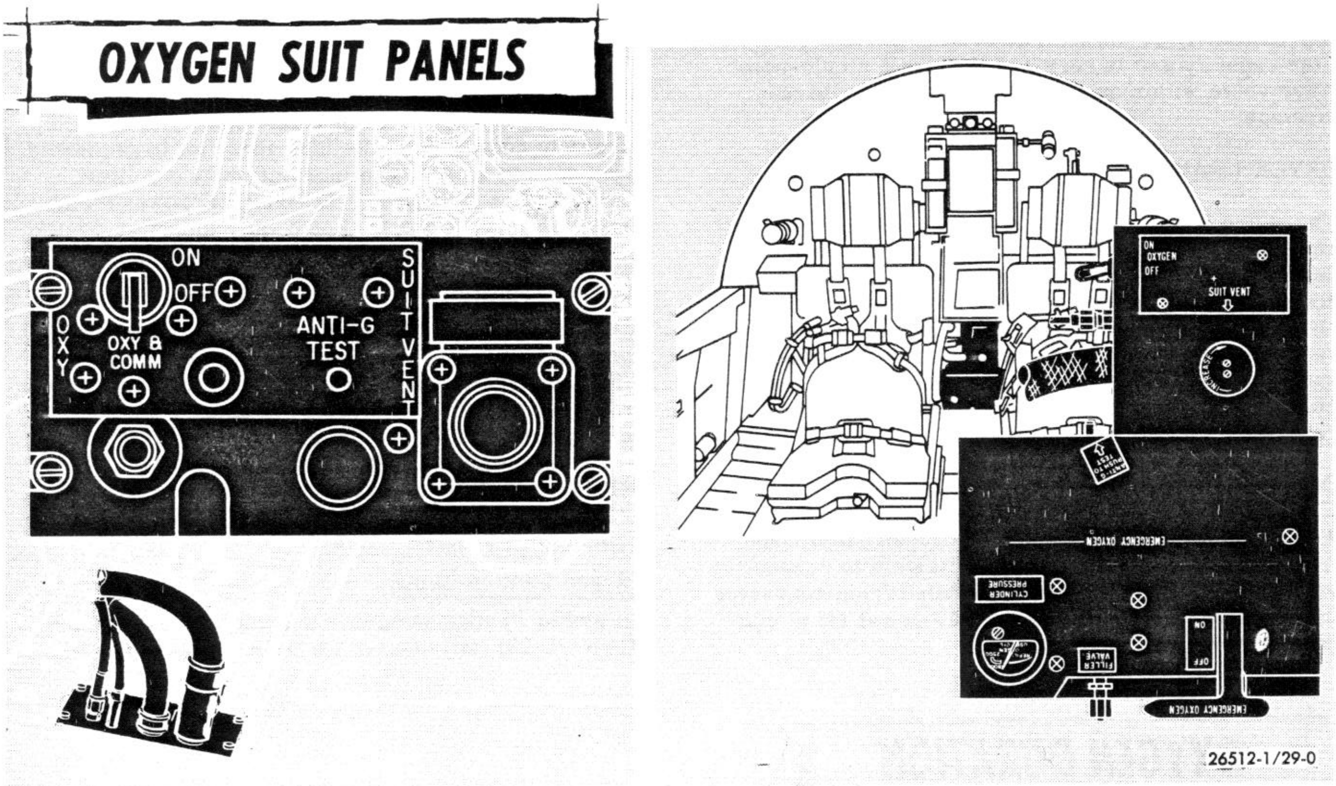


Figure 1-36

indicates 2 liters or less or when oxygen pressure is less than 42 (±2) psi. When caution lamp illuminates, inspection of oxygen quantity gage will determine whether lamp came on because of low quantity or low pressure. When lamp is illuminated, letters OXY will be visible. The oxygen caution lamp operates on 28-volt DC power from 28-volt DC essential bus.

EMERGENCY OXYGEN SYSTEM

The crew module is equipped with an emergency oxygen system consisting of two oxygen bottles, a pressure reducer, a pressure gage, and a manual handle. The system is activated automatically during ejection. Should the automatic feature fail, it can be activated by manual handle. During other phases of flight, this system provides oxygen if normal oxygen system fails or is depleted. When activated either manually or automatically, gaseous oxygen at 1800 to 2100 psi flows to a pressure reducer where it is reduced to 50 to 90 psi. It is then routed into normal oxygen system upstream of oxygen control valves. Sufficient emergency oxygen is available for 10 minutes duration at 27,000 feet cabin altitude.

Emergency Oxygen Handle

The emergency oxygen handle on MCO's oxygen panel (figure 1-36) is for manual activation of emergency oxygen system if automatic activation fails during crew module ejection sequence. Raising handle will open emergency oxygen pressure reducer, allowing oxygen to flow to each oxygen control valve.

Emergency Oxygen Pressure Gage

The emergency oxygen pressure gage on MCO's oxygen control panel (figure 1-36) indicates pressure in emergency oxygen bottles. The gage is marked REFILL in red region and FULL in black region with index marks at 1800 and 2500 psi.

LIGHTING SYSTEM

The lighting system is divided into exterior and interior lights over which pilot has complete control of selection, intensity, and mode of operation. All controls for exterior and interior lights are on lighting panel (figure 1-37). The landing/taxi light switch is on left console (figure 1-5).

EXTERIOR LIGHTING

The exterior lights include position lights, formation lights, anti-collision, and a landing/taxi light. The position lights consist of a red left wing tip light, a green right wing tip light and a white tail light. Supplemental position lights consist of a red light in left glove area and a green light in right glove area. The wing tip position lights will illuminate when the wing

sweep angle is between 16 and 30 degrees. When the arc swept aft of 30 degrees, the tip lights go out and the glove lights are illuminated. The reverse will occur as the wings are swept forward. The formation lights consist of two lights, on the upper and lower surfaces of each wing tip, one yellow light on the forward fuselage, and one yellow light on each side of the aft fuselage. The anti-collision lights consist of two rotating beacons, one on top and one on bottom of fuselage. A landing/taxi light is on nose landing gear.

INTERIOR LIGHTING

The interior lights include red instrument panel and console lights, red and white flood lights, and utility lights. The instrument panel and console lights consist of five circuits, with individual control selectors for flight instruments, engine instruments, left and center console, right console, and right main instrument panel. The flood lights consist of left, center, and right red flood lights, and high intensity white flood lights at various locations around cockpit. Red flood lights provide cockpit lighting should instrument panel and console lights fail. Each red flood light has an individual control selector. The white flood lights provide high intensity lighting to assist in preventing temporary blindness from lightning. One control selector adjusts intensity of all white flood lights. Two utility lights, one for each side of cockpit, provide individual work lights. They are normally stowed on left side of aft bulkhead and on right side wall, but can be moved to various locations about the crew station. The front of each utility light can be rotated to change color from white to red and vice versa. A rheostat on aft end of each light must be turned clockwise to turn light on and set desired intensity.

LIGHTING PANEL

All selectors for control of exterior and interior lights except for landing/taxi light are on lighting panel (figure 1-37). The lighting panel is on aft bulkhead console and can be reached by either crewmember.

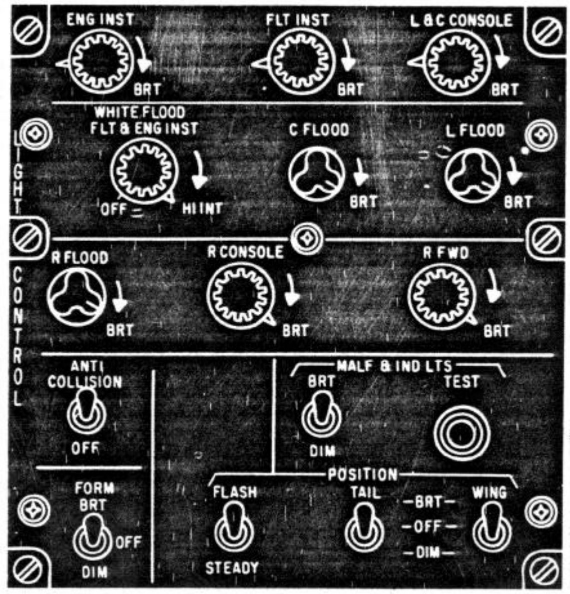
Position Light Switches

Illumination of position lights is controlled by three switches on lighting panel (figure 1-37). Two switches placarded WING and other placarded TAIL, have three positions marked BRT, OFF, and DIM, are for selecting desired intensity. A third switch with two positions placarded FLASH and STEADY controls operation of position lights. When FLASH is selected, position lights will flash at a rate of 80 cycles per minute.

Formation Lights Switch

A switch placarded FORM on lighting panel (figure 1-37) provides on/off selection and intensity control of formation lights. The switch positions are marked BRT (bright), OFF, and DIM.

LIGHTING PANEL



26512-1/42-0

Figure 1-37

Anti-collision Lights Switch

The anti-collision switch labeled ANTI COLLISION on lighting panel has one position marked OFF and an unmarked on position which is the up position. Placing switch to up will cause lights to illuminate and rotate.

Interior Lighting Selectors

Nine interior lighting control selectors on lighting panel control various internal lighting circuits. The full counterclockwise position of each selector turns lights off. As selectors are turned clockwise, detent positions at spaced intervals vary intensity of lights from off to full brightness. Five selectors control red instrument panel and console lighting. Selectors are labeled and control respective circuits as follows:

FLT INST	- Pilot's instrument panel
ENG INST	- Engine instruments
L & C CONSOLE	- Left and center consoles
R CONSOLE	- Right console
R FWD	- MCO's instrument panel

The red flood lights are controlled by individual selectors marked R FLOOD, C FLOOD, and L FLOOD for right, center, and left flood lights, respectively. A single selector marked WHITE FLOOD FLT & ENG

INST controls all white flood lights. This selector is similar to other selectors except it is marked OFF at the full counterclockwise position and HI INT (high intensity) near the full clockwise position. Turning selector past HI INT turns all white flood lights to maximum intensity.

LANDING AND TAXI LIGHT

The landing and taxi light switch labeled LDG/TAXI LT on left console (figure 1-5) has two positions marked ON and OFF. A limit switch on nose gear doors will turn off light if it is illuminated when gear is retracted.

MISCELLANEOUS EQUIPMENT

THERMAL RADIATION PROTECTION

Thermal radiation protection is provided by side curtains on canopy hatches and a hinged forward panel between glare shield and windshield.

LIQUID CONTAINERS

Two insulated liquid containers supply crew with hot or cold liquids during flight. The containers are stowed in recessed receptacles in seat bulkhead, outboard of each headrest. A spring-loaded latch on front of each receptacle holds respective container firmly in place against a coil spring in bottom of receptacle when container is stowed. Each container holds approximately 1 quart.

MIRRORS

Four rear view mirrors, two on each side of cockpit canopy frame, are installed to permit crew rearward vision without moving from their normal sitting position. The mirrors are adjustable in tilt only.

MAP STOWAGE

Two map cases are on left and right sidewalls. A nylon retaining strap, attached to each map case, extends upward, and attaches to cockpit sidewall fairing.

DATA STOWAGE CASE

A black nylon vinyl coated data case is in outboard aft end of right console. It consists of the case and a flap with a metal snap fastener to prevent data from inadvertently falling from case. The case is labeled DATA STOWAGE.

LETDOWN CHART STOWAGE

Two letdown chart stowage compartments are on each side of aft bulkhead console. The right compartment is labeled LETDOWN CHART HOLDER, left compartment is labeled LETDOWN CHARTS. Each compartment contains a strap and fastener to secure charts and holder.

EJECTION SYSTEM SAFETY PIN STOWAGE

A stowage compartment at aft end of left sidewall is used for stowing the ejection system safety pins.

CHECKLIST STOWAGE

A space for stowing checklist is provided on left sidewall. A nylon strap retains checklist in place.

FOOD STOWAGE COMPARTMENTS

A food stowage compartment is provided for crew on aft bulkhead aft of pilot's seat. The door of compartment is held closed by a spring-loaded latch.

CHART BOARD STOWAGE

Space is provided for MCO's stowage of a chart board on right side of seat. A fabric strap snaps over chart board when it is stowed to hold it securely in place.

HOOD STOWAGE COMPARTMENT

A hood stowage compartment on right side of aft bulkhead, just above MCO's relief container, is provided to store attack radar scope hood.

CREW ENTRANCE LADDERS AND STEPS

Crew entrance ladders and steps on each side of fuselage give crew access to crew module without aid of ground support equipment. When not in use, both sets of ladders and steps are retracted into sides of fuselage. Each left or right ladder and step can be electrically unlocked permitting extension from inside the crew module. Pushbutton releases are provided on outside of fuselage to manually unlock and permit extension of ladders and steps from the ground. The ladders and steps must be manually stowed from the ground.

ENTRANCE LADDER AND STEP SWITCH

The entrance ladder and step switch, labeled LADDER, on ground check panel, has three positions marked L, R, and OFF. Positioning switch to L or R will provide 28-volt DC power to a solenoid in respective ladder and step to release lock for extension. The switch is spring-loaded to center OFF position.

STANDBY COMPASS

A conventional standby compass is the center above MCO's instrument panel (figure 1-4). It is a semi-float-type compass suspended in compass fluid. A pair of magnets attached to compass card align with earth's magnetic field to present magnetic heading indications. Extraneous magnetic fields are minimized by built-in permanent magnets.

CLOCK

The clock on landing gear panel (figure 1-25) is self-contained and mechanically actuated. It is an 8-day clock and incorporates a 1-hour elapsed time capability. A winding and setting selector is in lower left corner of instrument face. The knob is turned in a clockwise direction to wind the clock, and when selector is pulled out it is used to set hour and minute hands. An elapsed time selector in upper right corner controls elapsed time mechanism. This mechanism starts, stops, and resets the sweep second and elapsed time hands.

part 3

AIRCRAFT SERVICING

TABLE OF CONTENTS

AIRCRAFT SERVICING

The following servicing data is provided to lend assistance if aircraft lands at a strange field or maintenance crews are unfamilar with the aircraft.

Fuel . . . . MIL-J-5624 (JP-5 or JP-4)

Hydraulic Fluid MIL-H-5606

Liquid Oxygen . . . . MIL-O-27210, type II

Oil MIL-L-23699

Windshield Wash Alcohol MIL-C-5543

Note

When operating in and out of military airfields, consult the current USAF/USN Enroute Supplement for compatible servicing units, fuel, etc.

AIRCRAFT REFUELING

Single-point refueling is provided for ground servicing and is accomplished through a standard ground refueling receptacle on the left side of fuselage. All tanks are equipped with refueling automatic shutoff valves. Service aircraft with specification MIL-J-5624, JP-5 or JP-4 fuel.

PNEUMATIC PRESSURES

Note

Dry nitrogen, specification BB-N-411, type 1, grade A, is preferred for charging pneumatic systems. Charge gas may be compressed air if nitrogen is not available.

Primary and Utility Hydraulic Reservoirs (2) . . . . 500 psi

Pneumatic Engine Inlet Reservoirs (2) . . . . 3,000 psi

Wheel Brake Accumulators (2) . . . . 800 psi

Alternate Landing Gear System . . . . 3,000 psi

Canopy Counterpoise System . . . . 648 psi

Overwing Fairing Pneumatic System . . . . . 1,800 psi

Damper Serve Accumulators (2) . . . . 1,400 psi

Nose Gear Shock Strut. . placarded on nose gear strut

Nose Gear Shock Strut Pneumatic Tank. . . . placarded on nose gear strut

Nose Wheel Tires (2) . . . 150 psi (land based) 400 psi (carrier based)

Main Gear Shock Struts (2) . . . . placarded on main gear strut

Main Wheel Tires (2) . . 205 psi (land based) 240 psi (carrier based)

Tailbumper Dashpot . . . . 57 ± 5 psi

Arresting Hook Dashpot . . . . 630 psi

RESERVOIRS

Primary Hydraulic . . . . Approx. 3 US gallons

Utility Hydraulic . . . . Approx. 9 US gallons

Engine Oil (2) . . . . 20 quarts

Constant Speed Drives (2) . . . . 10.8 pints

Liquid Oxygen Converters (2) . . . . 10 liters

Gaseous Oxygen System (Emergency) . . . . 1800 psi at 70° F (minimum)

Environmental Control System Water Tank . . . . 27 US gallons

Windshield Wash Tank (water alcohol) . . . . 2 quarts

Emergency Pressurization (High-Pressure breathable air) . . 3,000 psi at 70° F (minimum)

TURNING RADIUS AND GROUND CLEARANCE

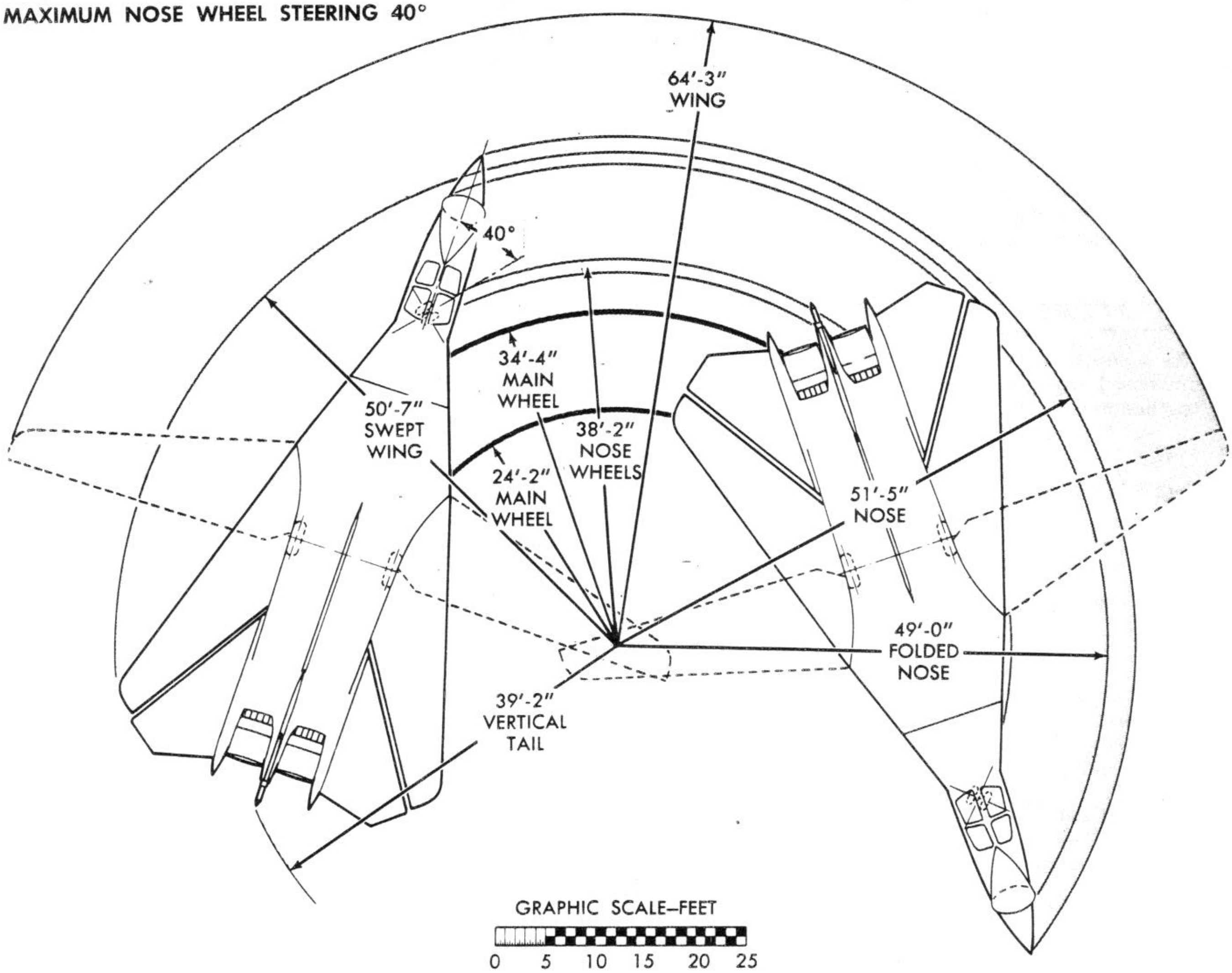
See figure 1-38.

DANGER AREAS

See figure 1-39.

TURNING RADIUS AND GROUND CLEARANCE

MAXIMUM NOSE WHEEL STEERING 40°



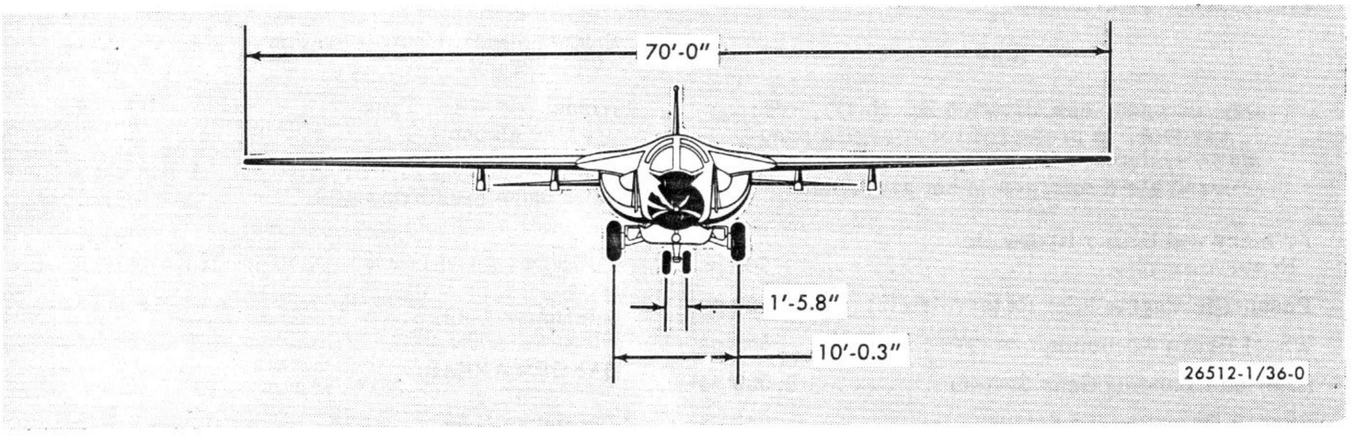


Figure 1-38

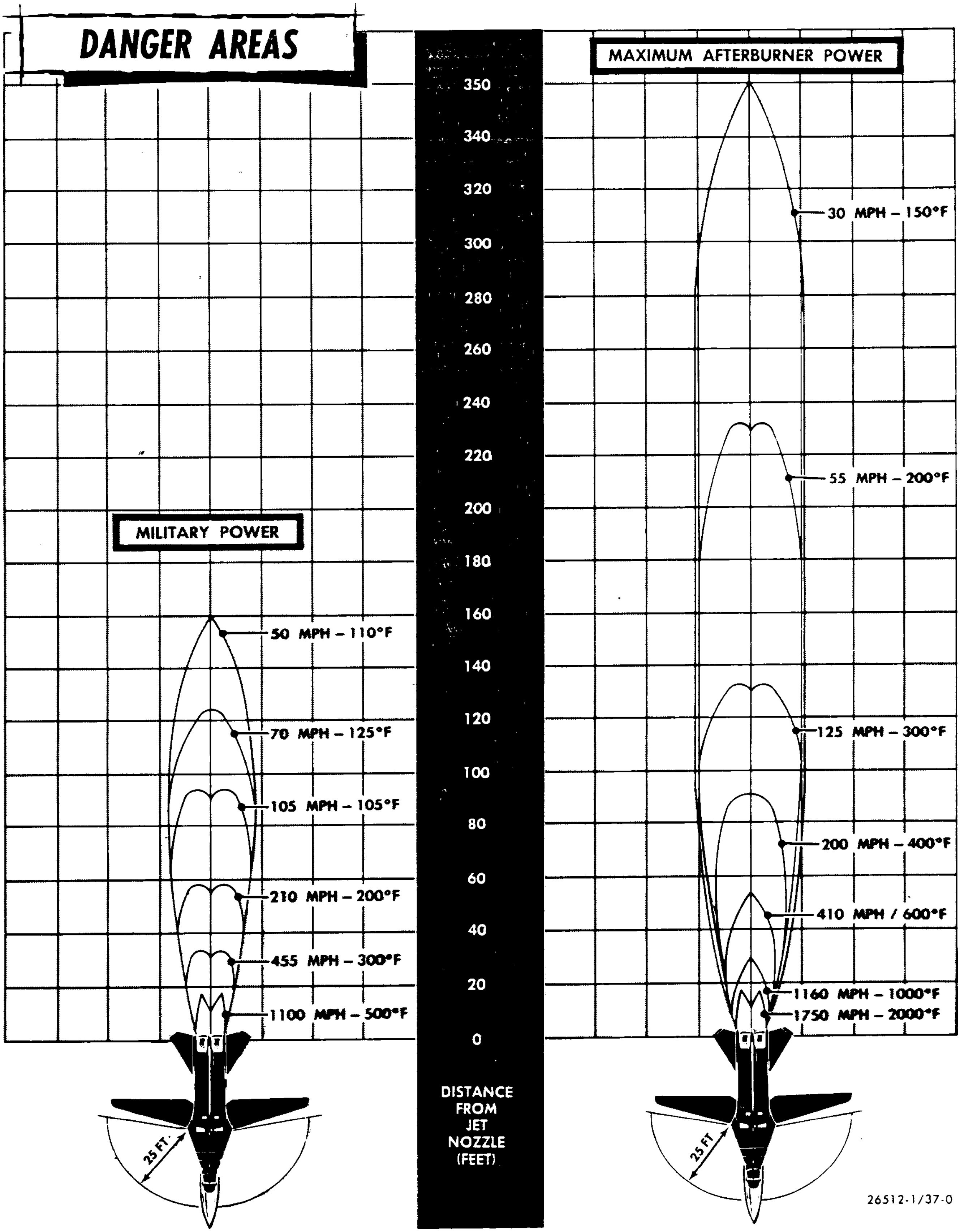

DANGER AREAS
MILITARY POWER
50 MPH – 110°F
70 MPH – 125°F
105 MPH – 105°F
210 MPH – 200°F
455 MPH – 300°F
1100 MPH – 500°F
25 FT.
350
340
320
300
280
260
240
220
200
180
160
140
120
100
80
60
40
20
0
DISTANCE FROM JET NOZZLE (FEET)
MAXIMUM AFTERBURNER POWER
30 MPH – 150°F
55 MPH – 200°F
125 MPH – 300°F
200 MPH – 400°F
410 MPH / 600°F
1160 MPH – 1000°F
1750 MPH – 2000°F
25 FT.
26512-1/37-0

Figure 1-39

part 4

AIRCRAFT OPERATING LIMITATIONS

TABLE OF CONTENTS

INTRODUCTION

This section includes limitations that shall be observed for safe and efficient operation of the engines and the aircraft. Instrument markings giving various operating limitations are shown in figure 1-40.

Note

The airspeed indicated on the airspeed mach indicator has been calibrated for pitot-static system errors by the CADC and is actually KCAS (knots calibrated airspeed). However, this airspeed is referred to as KIAS (knots indicated airspeed) throughout this manual since it is read directly from the instrument.

CAUTION

When any limits are exceeded, the flight crew shall log the condition on the yellow sheet (OP NAV FORM 3760-2) to ensure corrective action or appropriate inspection is accomplished.

The limitations contained in this section represent the most restrictive semi-permanent limitations currently in effect. Limitations of a temporary nature which are more restrictive than those included are contained in Temporary Flight and Operating Limitations Report For The F-111B (FZM-12-9220).

MINIMUM CREW REQUIREMENTS

A minimum crew requirement for safe operation is two.

ENGINE LIMITATIONS

Engine limitations are shown in figure 1-41.

AIRSPEED LIMITATIONS

MINIMUM FLIGHT AIRSPEEDS

Landing Gear and Flaps Up

The minimum permissible airspeeds with landing gear and flaps up are presented in figure 1-42 as a function of gross weight.

Gear and Flaps Down

The maximum allowable nose boom indicated angle-of-attack for 16 or 26 degrees wing sweep positions with landing gear and slats/flaps down is 21 degrees.

MAXIMUM FLIGHT AIRSPEEDS

The maximum permissible airspeeds are shown in figures 1-43 (Sheets 1, 2, and 3) and are tabulated below. These airspeeds are test nose boom indicated airspeeds based on measured airspeed position error.

Gear and Flaps Up:

Wing Sweep = 16 degrees . . . . 360 KIAS/0.77 M
Wing Sweep = 26 degrees
S. L. to 17,500 feet . . . . 0.77 M
17,500 feet to ceiling . . . . . . 380 KIAS/0.91 M
Wing Sweep = 45/50 degrees . . . . 0.86 M
Wing Sweep = 72 degrees . . . . See figure 1-43 (Sheet 3)

Gear Up or Down, Flaps 30 Degrees or More

Wing Sweep = 16/26 degrees . . . . 220 KIAS

Gear Up or Down, Flaps Less than 30 Degrees

Wing Sweep = 16/26 degrees . . . . 240 KIAS

Landing Gear Operation

Extension, Retraction or Flight With Gear Down . . . . 285 KIAS

CAUTION

For emergency landing gear extension sweep wings to 16 degrees, extend flaps to 35 degrees, and decelerate to 140 KIAS prior to extending gear.

Speed Brake Operation

Do not extend speed brake above the following limits:

16/26 degrees wing sweep . . . 440 KIAS/.77 M

45/50 degrees wing sweep	440 KIAS/.84 M
72 degrees wing sweep	440 KIAS/1.2 M

RAM Air Scoop Operation

Do not extend the RAM Air Scoop above 320 KIAS.

Crew Module Ejection Limitations

Recommended limit speed for ejection is 700 KIAS.

GROUND OPERATION SPEEDS

Nose and Main Gear Limits

The design tire speed is 157 knots ground speed. Take-off or landing speeds in excess of this speed shall be noted.

Taxi Limits

Lateral tip over of the airplane may occur at high taxi speeds. Refer to figure 1-44 for allowable rudder deflection as a function of taxi speed.

Brake Application Limits

Brake application speed limitations are presented in figure 1-45 as a function of gross weight. If maximum braking capacity is utilized (Danger Zone), wheel blow out plugs will relieve tire pressure within 3 to 15 minutes after the aircraft has stopped and provisions should be made to cope with possible fires which may start shortly after blow-plug release.

Note

If a maximum effort landing is made, do not attempt to takeoff until brakes have cooled to 38° C (100° F).

Anti-Skid System

Minimum skid control speed is 20 KIAS. Employ light to medium braking below this speed.

MANEUVER LIMITATIONS

ROLLING MANEUVERS

With either the landing gear or flaps extended or both, 1.0g 45 degrees normal force detent banks are permitted. With gear and flaps up, normal force detent (1/2 stick) rolls are restricted to the bank angle limitations of figure 1-43 (Sheets 1, 2 and 3).

CAUTION

When carrying wing fuel at less than 45 degrees wing sweep do not exceed 1/2 force detent (1/4 stick) above 440 KIAS.

Note

The force detent is located at approximately one-half of the available 4.75 inches of lateral stick deflection.

SIDESLIP MANEUVERS

In the power approach configuration or in the partial flap configuration, the following sideslip angles or the rudder deflection of figure 1-46 is permitted, whichever is less:

0 to 10 degrees angle-of-attack	±20 degrees β
10 to 15 degrees angle-of-attack	±15 degrees β

With gear and flaps up; ± 7-1/2° β sideslips are permitted up to 0.77 M for all wing sweeps. ±3° β sideslips are permitted between 0.77 M and 1.6 M. No intentional sideslips above 1.6 M.

ANGLE-OF-ATTACK

The following are the angle-of-attack limitations for the C_R & PA configuration:

Gear and Flaps Down	21 degrees angle-of-attack
Gear and Flaps Up	
16 to 45 degrees wing sweep	-3 to 16 degrees angle-of-attack
46 to 72 degrees wing sweep	-5 to 23 degrees angle-of-attack

WING SWEEPING

Gear and Flaps Extended

Wing sweep maneuvers with gear and flaps extended are permitted from 0.5g to 2.0g at gross weights of 60,800 pounds or less. At gross weight above 60,800 pounds, see figure 1-47 for allowable load factors.

Gear and Flaps Retracted

Wing sweeping with the gear and flaps retracted is permitted from 0 to 2.0g.

CAUTION

Avoid use of lateral control during wing sweeping maneuvers.

PROHIBITED MANEUVERS

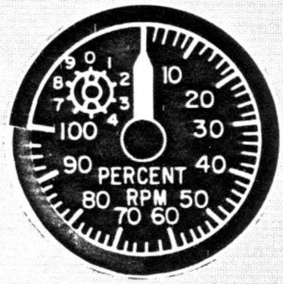
1. Stalls
2. Intentional spins
3. Operation of weapons bay door in flight
4. Large abrupt control inputs (excluding small displacement stick jabs and rudder kicks).
5. Catapult launches and arrested landings.

ACCELERATION LIMITATIONS

GEAR AND FLAPS EXTENDED

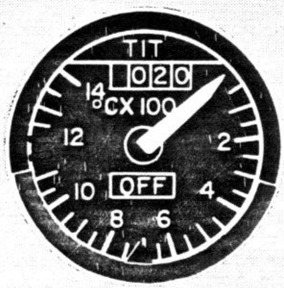
The allowable load factors with the gear up or down, and flaps extended are -0.5g to +2.0g at gross weights of 60,800 pounds or less. At gross weights above 60,800 pounds, refer to figure 1-47 for allowable load factors.

INSTRUMENT MARKINGS



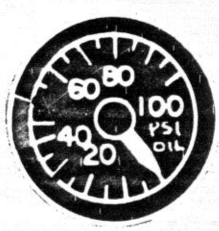
TACHOMETER

NORMAL OPERATING RANGE-PERCENT	64 TO 99.8
MAXIMUM OPERATING SPEED — PERCENT	99.8



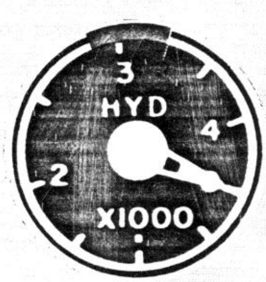
TURBINE INLET TEMPERATURE

NORMAL OPERATING RANGE	300 TO 1160°C
▲ STARTING (MOMENTARY)	705°C
MAXIMUM (MILITARY/AFTERBURNING)	1160°C
NORMAL RATED (MAXIMUM CONTINUOUS)	1015°C
ACCELERATION (ENGINE TRANSIENT)	1188°C



OIL PRESSURE

NORMAL RANGE	40 TO 50 PSI
MAXIMUM	50 PSI
MINIMUM (IDLE)	30 PSI



HYDRAULIC PRESSURE

NORMAL RANGE	2950 TO 3250 PSI
MAXIMUM	3250 PSI

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Figure 1-40

ENGINE OPERATING LIMITS

TF30-P-12 ENGINE

OIL: MIL-L-7808D
OR MIL-L-23699

FUEL: MIL-J-5624
GRADE: JP-4 OR JP-5

OPERATING CONDITIONS		OPERATING LIMITS		
THRUST SETTING	TIME LIMIT (MINUTES)	MAXIMUM MEASURED TURBINE-INLET TEMPERATURE (C°)	OIL PRESSURE (PSIG) NORMAL	OIL TEMPERATURE RANGE (°C)
MAXIMUM (AFTERBURNING)	45	1160°	45 ±5	40° – 120°
PARTIAL AUGMENTATION	45	1160°	45 ±5	40° – 120°
MILITARY	45	1160°	45 ±5	40° – 120°
NORMAL RATED	CONTINUOUS	1015°	45 ±5	40° – 120°
IDLE	CONTINUOUS	—	30 MINIMUM	40° – 120°
STARTING	MOMENTARY	705°	—	—
ACCELERATION	2 MINUTES	1188°	45 ±5	—

NOTE

THE TWO-MINUTE ACCELERATION TIME LIMIT COMMENCES WHEN THE THROTTLE IS FIRST ADVANCED.

RPM LIMITS

ANY OVERSPEED LIMIT EXCEEDED SHOULD BE REPORTED AS A DISCREPANCY AND MAXIMUM RPM NOTED.

OPERATING CONDITION	OPERATING LIMITS
IDLE	64–73 PERCENT RPM
MAXIMUM SPEED N_2	14,550 RPM – 99.8 PERCENT
OVERSPEED, ENGINE INSPECTION REQ'D.	TO BE SUPPLIED WHEN AVAILABLE
OVERSPEED, ENGINE CHANGE REQUIRED	TO BE SUPPLIED WHEN AVAILABLE

26512-1/127-0

Figure 1-41

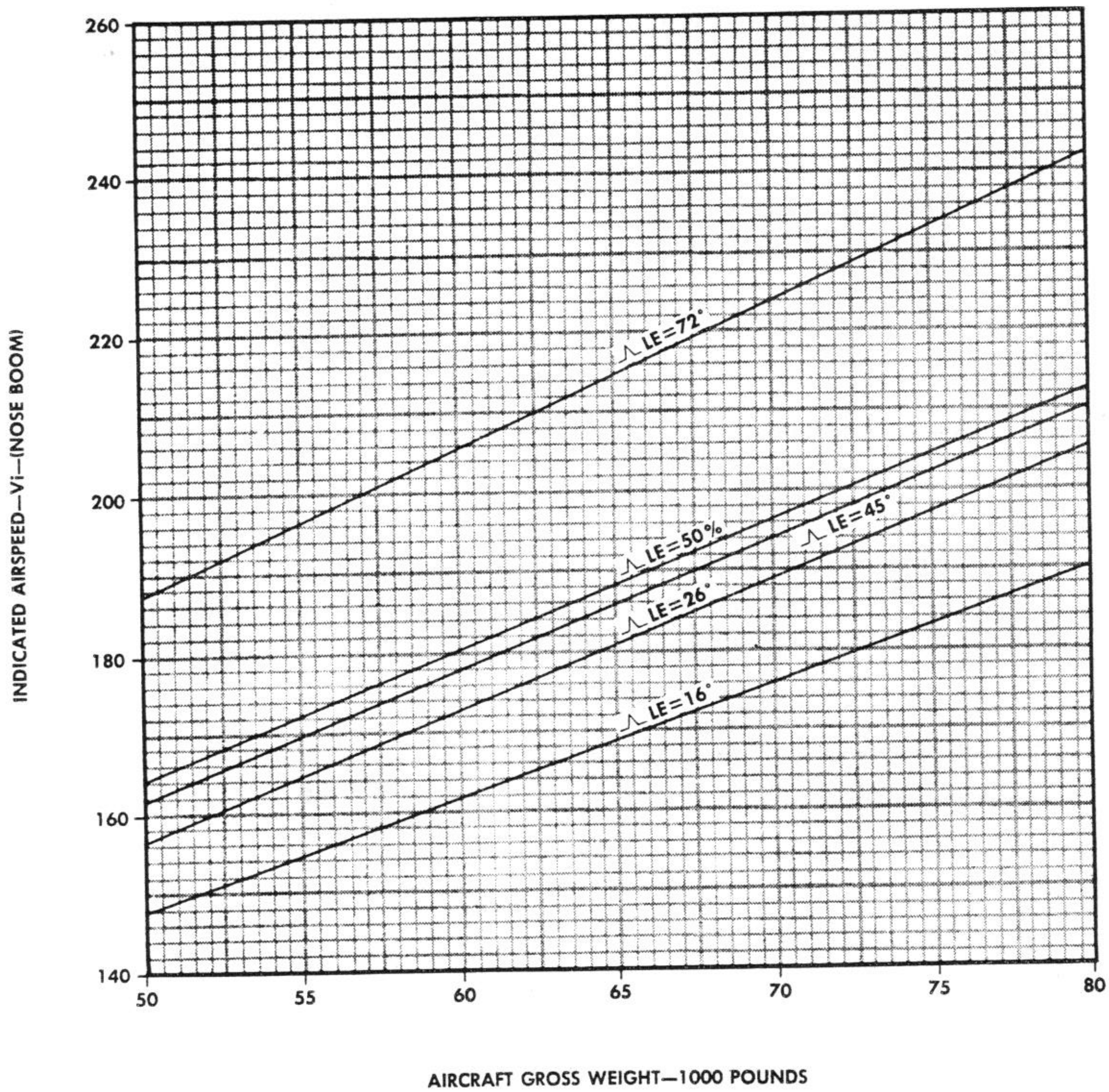
MINIMUM RECOMMENDED FLYING SPEEDS

GEAR AND FLAPS UP

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/119-0

Figure 1-42

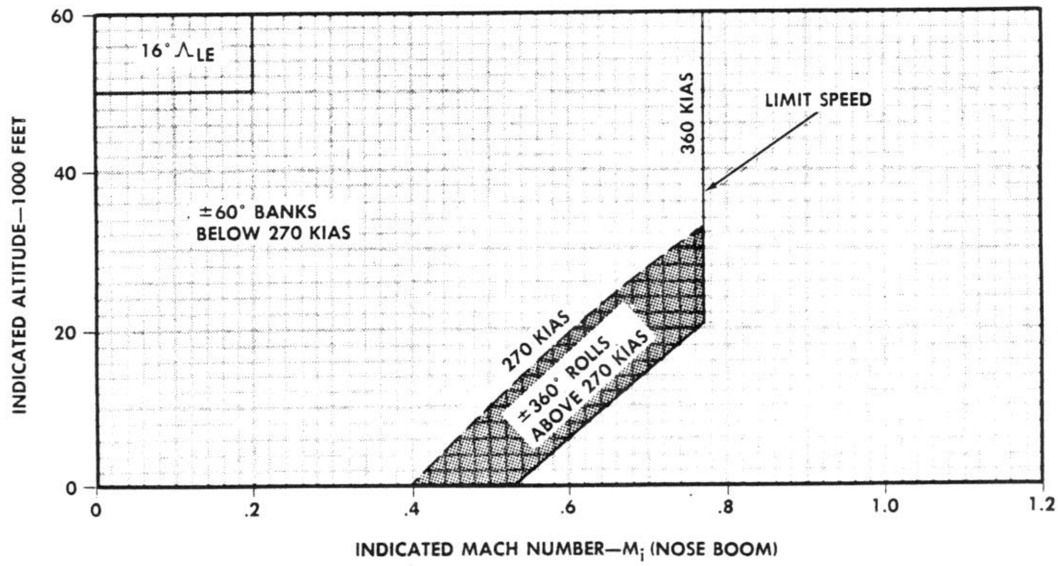
SPEED AND ROLLING LIMITATIONS

GEAR AND FLAPS UP

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



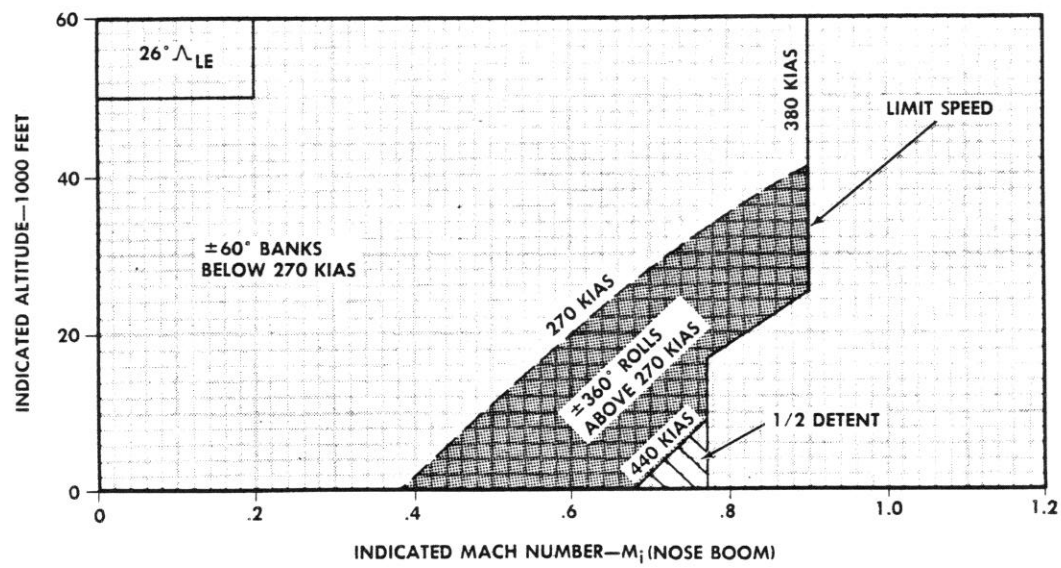


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Figure 1-43 (sheet 1)

SPEED AND ROLLING LIMITATIONS

CLEAN, GEAR AND FLAPS UP

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

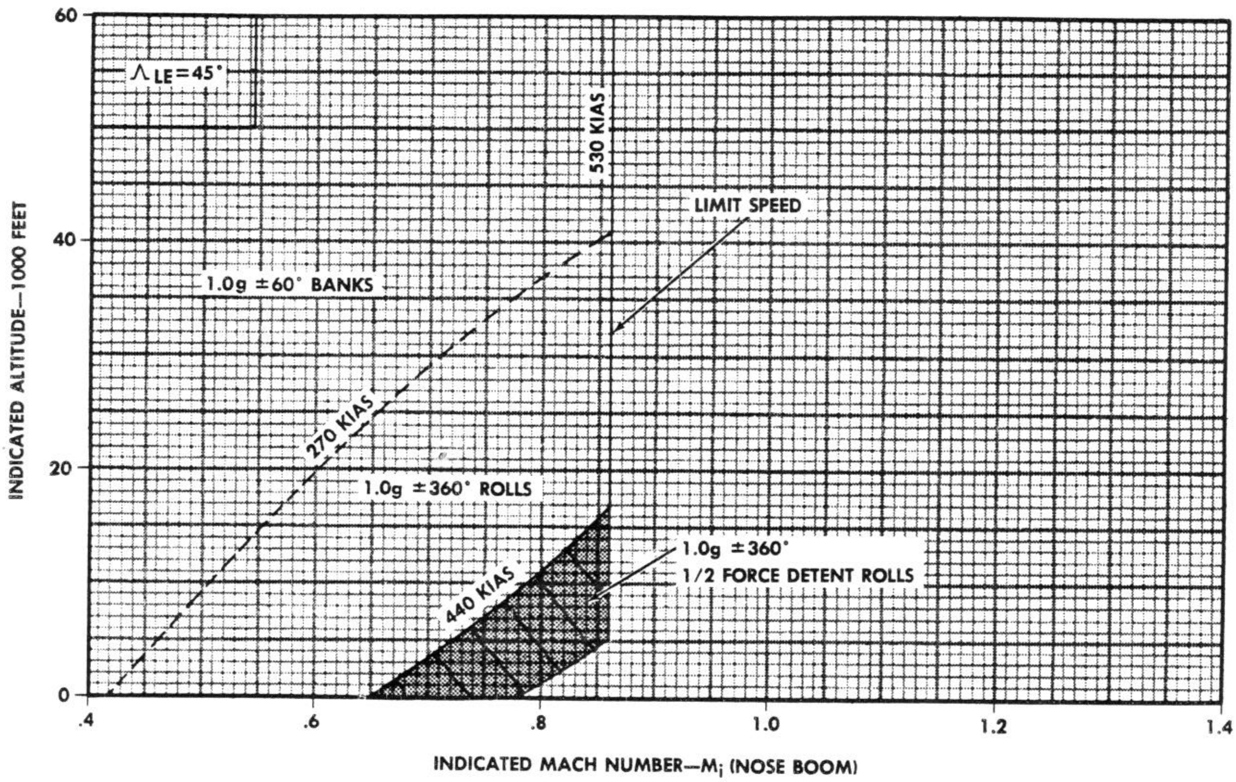

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Figure 1-43 (sheet 2)

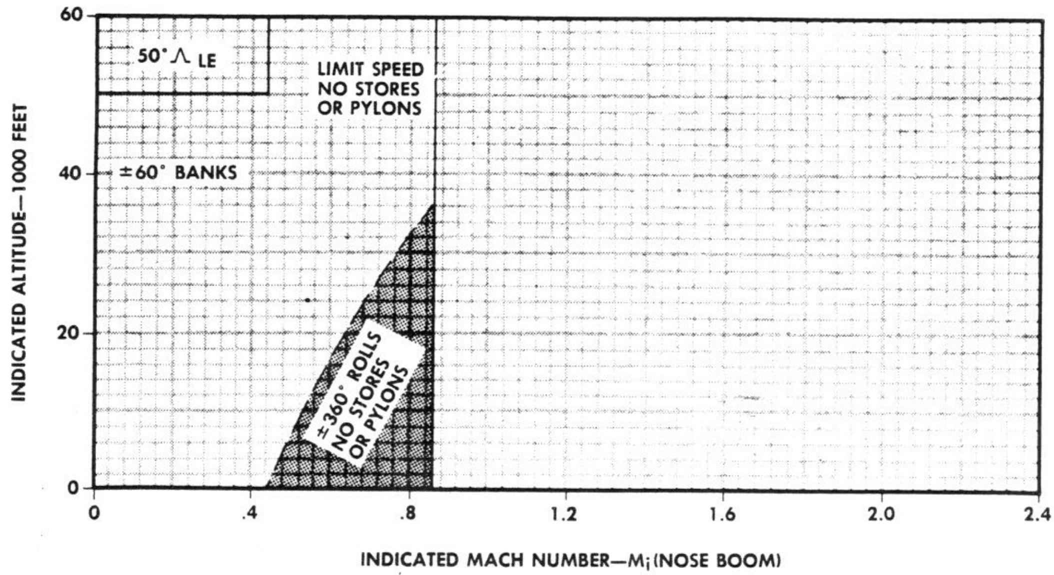
SPEED AND ROLLING LIMITATIONS

GEAR AND FLAPS UP

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINES(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



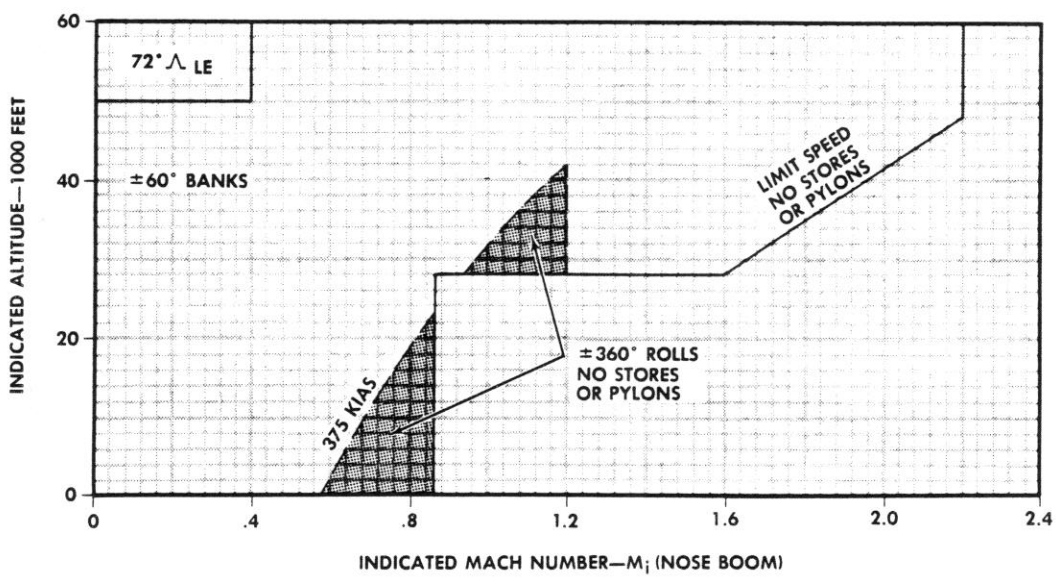


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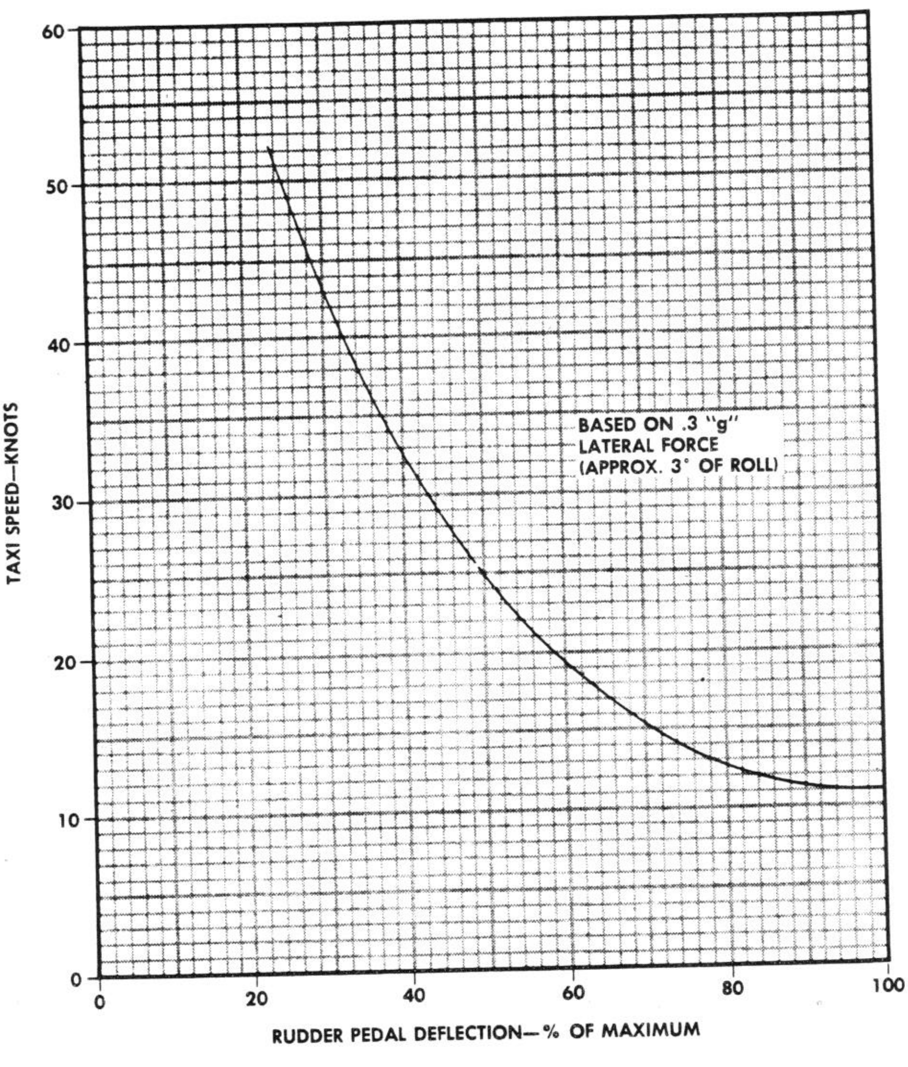
Figure 1-43 (sheet 3)

MAXIMUM PERMISSIBLE RUDDER DEFLECTION DURING TAXI

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/121-0

Figure 1-44

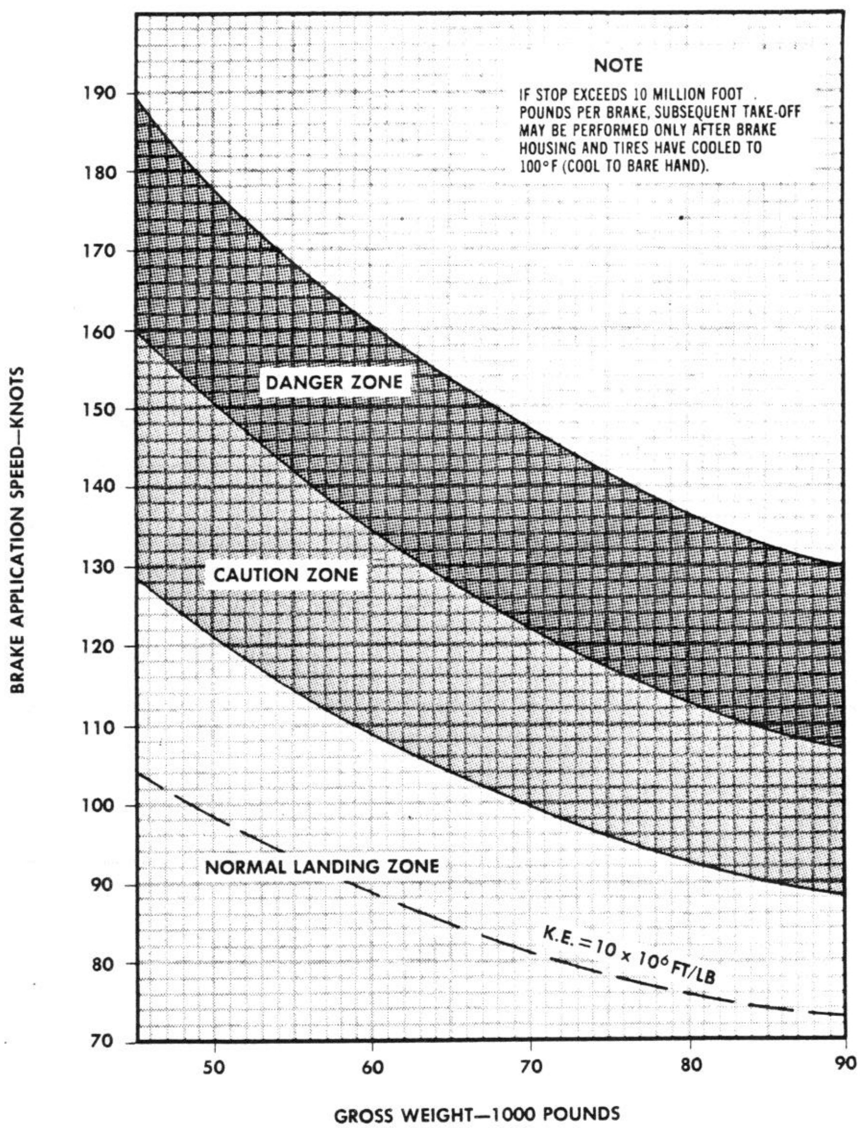
MAXIMUM SPEEDS FOR BRAKE APPLICATION

AIRCRAFT CONFIGURATION:
CL09 603—3 WHEEL
CL09 603—2 BRAKE

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/122-0

Figure 1-45

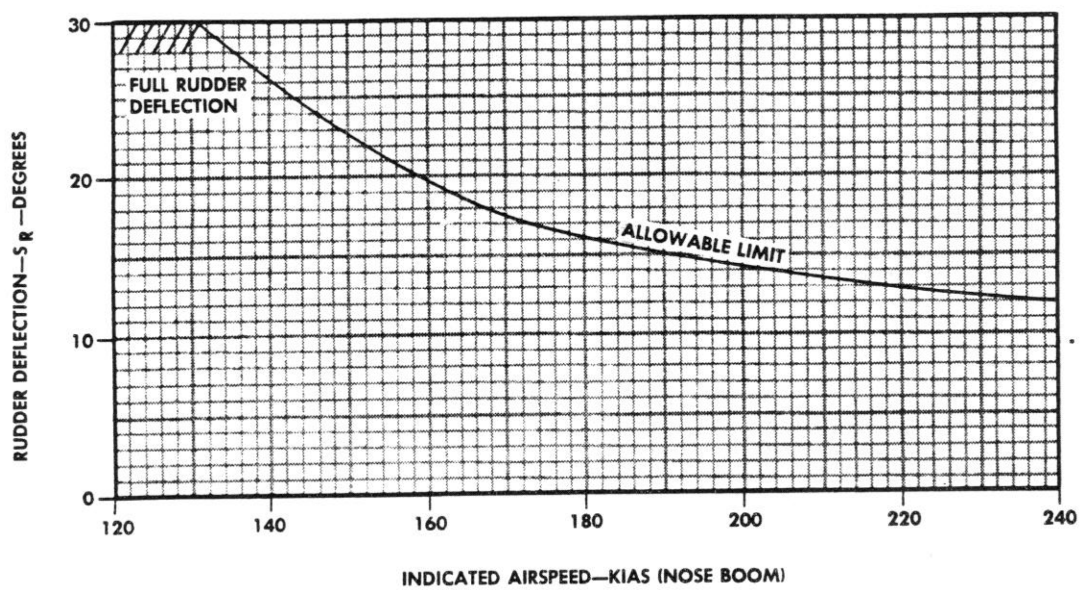
MAXIMUM PERMISSIBLE RUDDER DEFLECTION

GEAR DOWN

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/123-0

Figure 1-46

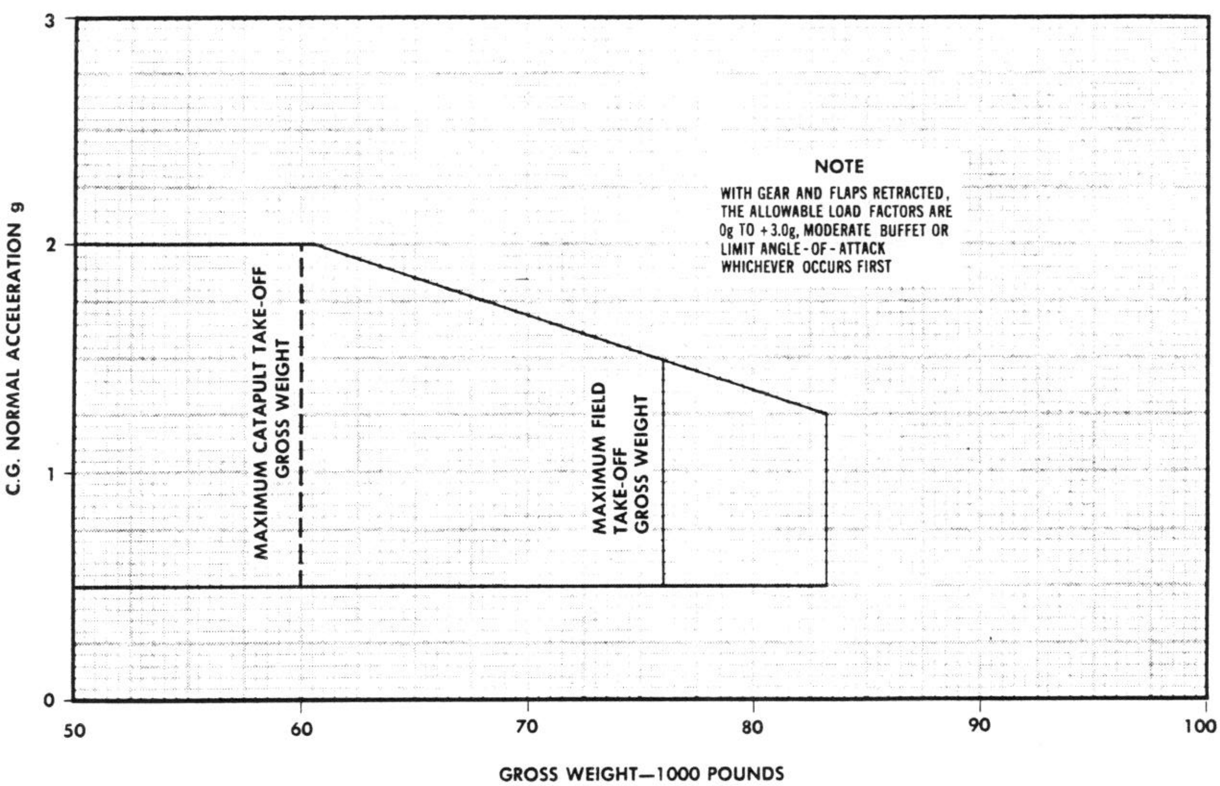
LOAD FACTOR—GROSS WEIGHT LIMITATION

GEAR AND FLAPS EXTENDED

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINES(S): (2) TF-30-P-12

FUEL GRADE: JP—5
FUEL DENSITY: 6.8 LB/GAL.



26512-1/124-0

Figure 1-47

CAUTION

Gear extension and retraction should be made in 1.0g level flight.

GEAR AND FLAPS RETRACTED

The allowable load factors with gear and flaps retracted are 0g to +3.0g or moderate buffet or angle-of-attack limits (whichever occurs first).

GROSS WEIGHT LIMITATIONS

The maximum recommended gross weights are as follows:

Taxi and field takeoff	76,000 pounds
Field landing (no flaps - sink rate not to exceed 960 feet per min.)	57,000 pounds
Above 57,000 pounds only normal flared landings up to	68,000 pounds

CENTER OF GRAVITY LIMITATIONS

WARNING

The crew module is not safe for ejection without its full crew and complement of survival equipment, or the equivalent ballast to maintain center-of-gravity within prescribed limits. Personal belongings or additional heavy equipment shall not be carried in the cockpit without compensation for CG or overweight effect. To assure stability of the module in the event of ejection, it must be loaded in accordance with TO 1-1B-40.

GEAR AND FLAPS EXTENDED

The aft center of gravity limits with the gear down and 35 degrees flaps are presented in figure 1-48 (Sheet 1) as a function of wing sweep angle.

CAUTION

- Lateral-directional maneuvering is not permitted aft of 42.0%, at 16 degrees wing sweep due to lack of available elevator control.
- Based on available elevator control, the forward center of gravity limit for landing at 26 degrees wing sweep is 42.0% MAC.

AFT CENTER OF GRAVITY LIMITS

The following are the aft center of gravity limits for flap settings less than 35 degrees.

WING SWEEP	LSTC ON	LSTC OFF
16 degrees	42.0% MAC	37.5% MAC
26 degrees	50.0% MAC	50.0% MAC

Gear and Flaps Retracted

Figure 1-48 (Sheet 2) presents the aft center of gravity limit for the 16 degree wind sweep position as a function of Mach number and the aft center of gravity limits for 26 to 72 degrees as a function of wing sweep.

COCKPIT PRESSURE LIMITS

In order to maintain cabin pressurization within the current limits, all flight operation shall be conducted with the cockpit pressurization selector switch in the COMBAT position.

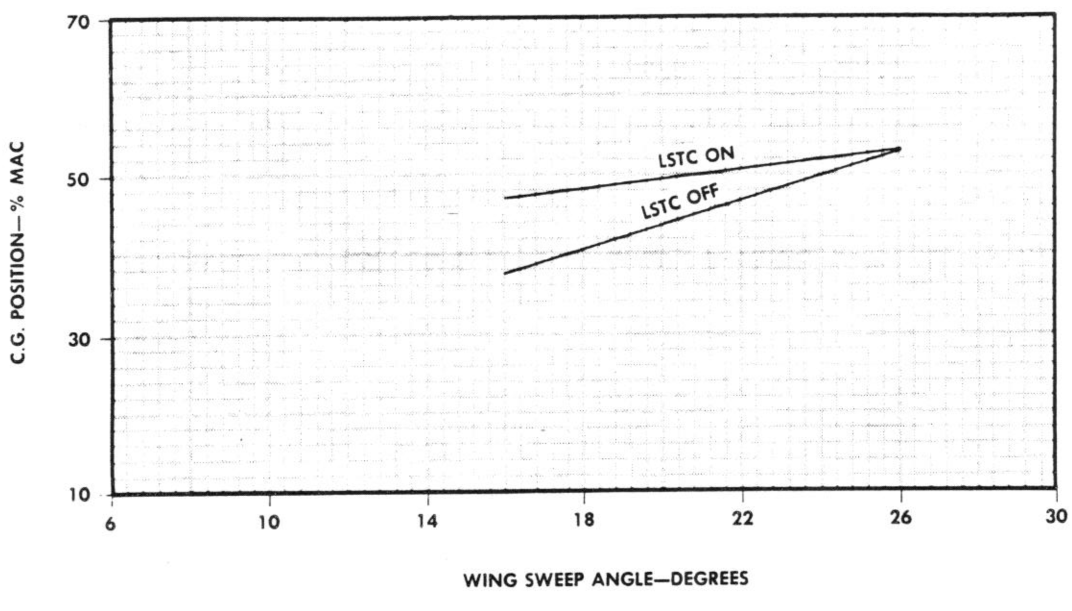
AFT CENTER OF GRAVITY LIMITATIONS

GEAR AND FLAPS DOWN ≥ 35°

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/125.1-0

Figure 1-48 (sheet 1)

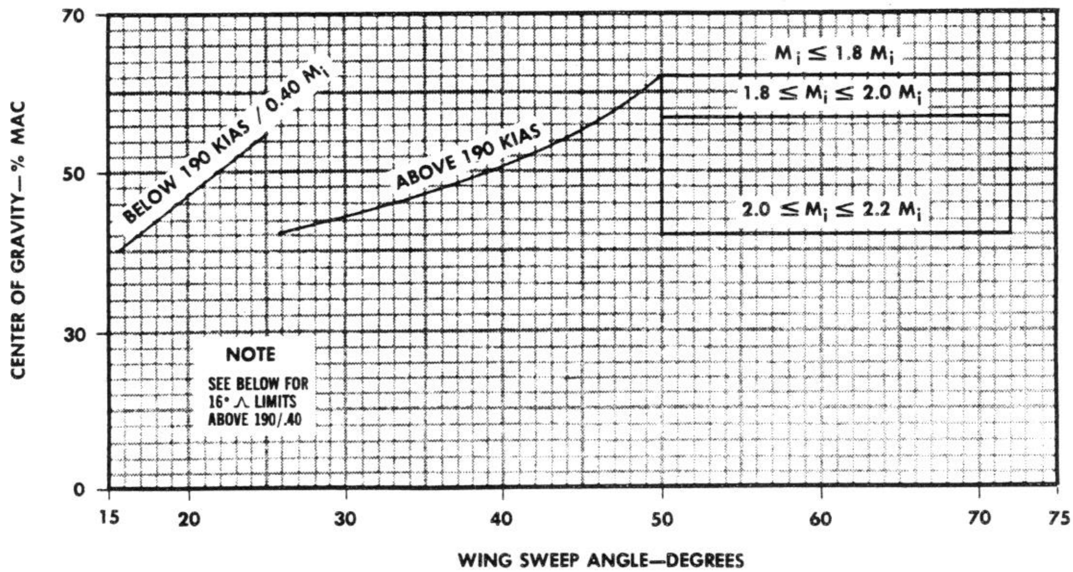
AFT CENTER OF GRAVITY LIMITATIONS

GEAR AND FLAPS UP—26°-72° WING SWEEP

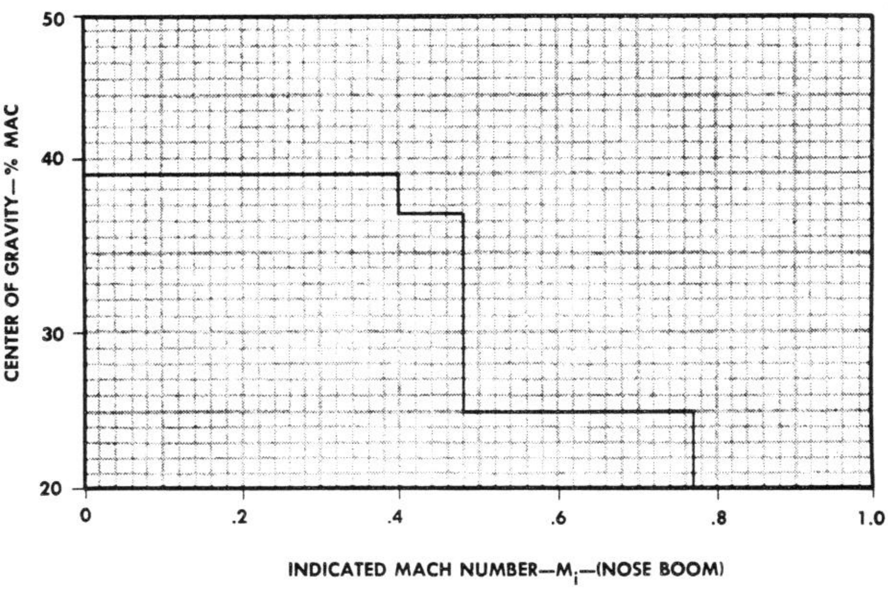
DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF-30-P-12

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



GEAR AND FLAPS RETRACTED—16° WING SWEEP



26512-1/125.2-0

Figure 1-48 (sheet 2)

section II
INDOCTRINATION

26512-1/58-0

TO BE SUPPLIED AT A LATER DATE

section III
NORMAL PROCEDURES

part 1
BRIEFING / DEBRIEFING

part 2
MISSION PLANNING

part 3
SHORE-BASED PROCEDURES (PILOT)

part 4
CARRIER-BASED PROCEDURES (PILOT)

part 5
SHORE-BASED PROCEDURES (MCO)

part 6
CARRIER-BASED PROCEDURES (MCO)

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part 1

BRIEFING / DEBRIEFING

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BRIEFING/DEBRIEFING RESPONSIBILITIES

The Commanding Officer shall ensure that every flight crew is properly and adequately briefed on all facets of the assigned mission. Through the Operations Officer, he shall assign the appropriate personnel to brief each flight, according to the mission and anticipated conditions. To this end, the following persons shall be responsible:

OPERATIONS OFFICER

The Operations Officer shall be responsible to the Commanding Officer for ensuring that appropriate personnel are assigned to conduct specified portions of the briefing for each flight.

BRIEFING OFFICER

The Briefing Officer assigned on the Flight Schedule shall have the overall responsibility for coordinating and conducting a proper and adequate briefing of the flight. This responsibility will be assumed by the flight leader if a Briefing Officer has not been assigned.

SQUADRON DUTY OFFICER

The Squadron Duty Officer shall be responsible through the Operations Officer for:

1. Ensuring that flight personnel are equipped with proper flight clothing, navigational kits, flight packets, flight plans, and survival equipment as necessary.
2. Providing the latest weather information, and advising the flight of NOTAMS and exercises in the operating areas which might jeopardize the safety of the flight.
3. Ensuring that other personnel required for specific briefings, such as AIO, LSO, Ordnance Officer, and Meteorologist, are present.
4. Keeping himself informed of the status of all aircraft, and assigning aircraft capable of performing the scheduled mission.

FLIGHT LEADER

The Flight Leader shall ensure that all members of his flight have received an adequate and proper briefing. He shall supplement each briefing as necessary.

PILOT-IN-COMMAND

Each Pilot-in-Command shall ensure that an adequate briefing has been obtained, and that his crew is fully prepared for the scheduled mission.

NON-OPERATIONAL BRIEFING

The Flight Leader may conduct briefings on training, familiarization, and other similar flights where only NOTAMS, weather, and communications information are required. Air intelligence, navigation, communications, and other cognizant officers will ensure that information for each briefing is current and readily available to the flight leader.

OPERATIONAL BRIEFING

Flight crews shall be given complete, comprehensive briefings on all operations. The Briefing Officer shall work in conjunction with the Operations Officer, Air Intelligence Officer, and other officers concerned, in preparing the necessary information. He shall make optimum use of all graphic presentation devices, maps, charts, etc., which are available.

PRE-OPERATIONAL BRIEFING

Immediately prior to all operations of appreciable complexity and duration, a general information briefing shall be given to familiarize personnel with the overall nature of the operation. The following topics shall be included:

1. The mission and objective of the operation, and the part the squadron will play in carrying them out.
2. A brief chronological breakdown of how the operation will be conducted.
3. The geographical area in which the operations will be conducted.

4. The forces involved, both friendly and enemy, and how they will be deployed.

5. The rules of engagement set down by the governing operation order.

6. Search and rescue, EMCOM, and other special communications procedures which will be used, including explanation of shackles and authenticators.

7. A discussion by the Briefing Officer of the principal attack tactics to be employed.

PRE-FLIGHT BRIEFING

These briefings are presented immediately before the launching of scheduled flights, and therefore must be carried out in the most expeditious manner. It is imperative that all pilots and missile control officers be in flight gear and ready for the briefing at the designated time. The briefing shall include, but not be limited to, the following:

Note

Information marked with an asterisk (*) shall be displayed on a status board in the briefing or ready room, and should be copied by pilots before commencement of the briefing.

1. Scheduling
   a. Event Number*
   b. Takeoff, recovery times, and recovery order*
   c. Aircraft-pilot lineup*
   d. Mission assigned to each aircraft*
   e. Marshal information
2. Mission
   a. CAP station assignment and control*
   b. Target aircraft rules of engagement
   c. Ground target description and procedures
3. Communications
   a. Channels and frequencies*
   b. Navigational aids*
   c. Lost communications procedures
   d. Reports required
   e. Authenticators, IFF
   f. EMCOM conditions
4. Participating Units
   a. Voice calls and side numbers*
   b. Disposition
   c. Utilization
   d. Friendly subs and surface units
5. Operations
   a. Instructions for coordinating other units
6. Ordnance
   a. Ordnance carried*
   b. Restrictions on use
7. Weather - Base, enroute, target, and divert field
   a. Wind: direction and velocity at surface and at applicable altitudes*
   b. Cloud coverage: present and forecast*
   c. Visibility*
   d. Sea state*
   e. Water and air temperature (cold weather)*
   f. Target weather*
   g. Divert weather*
8. Miscellaneous
   a. Other units in the area
   b. Restricted or danger areas
   c. Current NOTAMS, bulletins, and safety-of-flight information
   d. Flight leader brief on takeoff, rendezvous frequency switch, landing procedures, etc.
   e. SAR - participating units and procedures
   f. BINGO fuel
   g. Tanker aircraft information

part 2
MISSION PLANNING

TO BE SUPPLIED AT A LATER DATE

part 3
SHORE-BASED PROCEDURES (PILOT)

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PREPARATIONS FOR FLIGHT

A thorough briefing shall be conducted concerning normal and emergency procedures, crew coordination, and cooperation through the planned and alternate missions. For aircraft and crew module loading information, refer to Handbook of Weight and Balance.

Note

The crew module should not be considered flyable without its full crew and complement of survival equipment, or the equivalent ballast to maintain center-of-gravity within prescribed limits. Personal belongings or additional heavy equipment shall not be carried in the cockpit without compensation for CG or over-weight effect. To assure stability of the module in event of ejection, it must be loaded in accordance with TO 1-1B-40.

FLIGHT RESTRICTIONS

Refer to Section I, Part 4, Aircraft Operating Limitations, and Section XI, Performance (Charts) Data.

FLIGHT PLANNING

Refer to Section XI, Performance Data, to determine fuel consumption, airspeeds, power settings, and altitude required for the intended flight mission.

WEIGHT AND BALANCE

Refer to Section I, Part 4, for Aircraft Operating Limitations.

For loading information, refer to the Manual of Weight and Balance Data, AN 01-1B-40.

CHECK LISTS

The placarded takeoff and landing check lists (figure 3-1) on the pilot's instrument panel are used only as a reminder and list, in general terms, what to check. The pilot must be thoroughly familiar with the complete procedures outlined in this manual in order to know how these items should be checked. Each step within the check list must be performed or carried out in sequence as outlined in the appropriate check list and NATOPS Pocket Check List.

NORMAL ENTRANCE

See figure 3-3 for normal entry into the module cockpit via the boarding ladder.

TAKE-OFF AND LANDING CHECK LIST

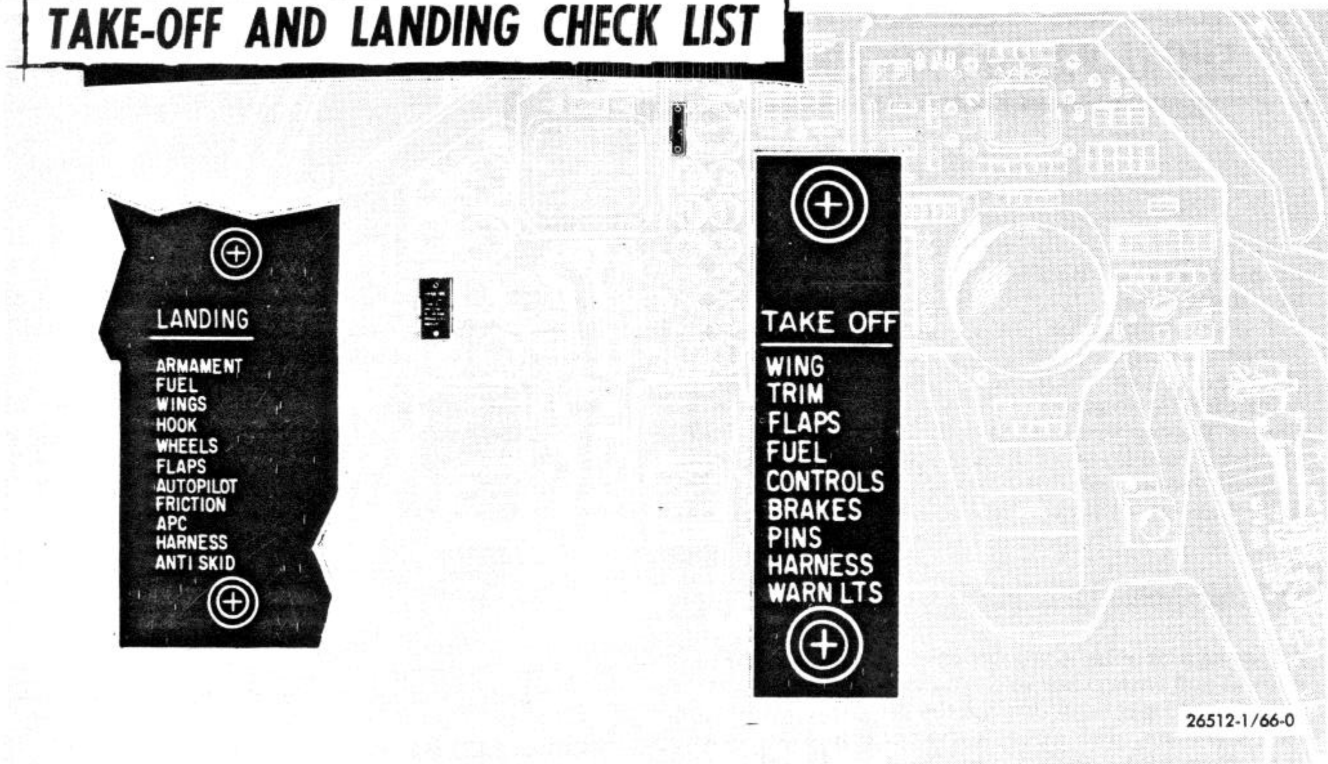


Figure 3-1.

PREFLIGHT CHECK

BEFORE EXTERIOR INSPECTION

OPNAV 3760-2 (Yellow Sheet) - CHECK
Check yellow sheet to determine the flight status and servicing of the aircraft.

EXTERIOR INSPECTION

Perform the exterior inspection as outlined in figure 3-2. Check all surfaces for any type of damage or fluid leaks that may have developed since the preflight inspection. Check all access doors and covers for security.

Note

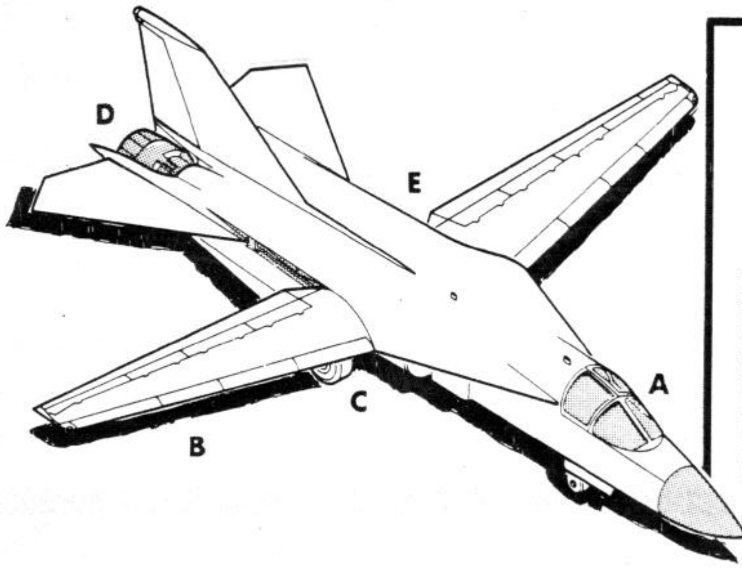
With the aircraft parked in alignment with the prevailing wind, it is not unusual for the engine fan and compressor to windmill. This rotation will cause no damage.

Ⓐ FORWARD FUSELAGE

1. Static ports - CHECK (L)
2. ADD probe - CHECK (L)
   Check for freedom of movement.
3. Beta probe - CHECK (L)
4. IR pod - CHECK (L)

CONTINUED ON NEXT PAGE

EXTERIOR INSPECTION



GENERAL CHECKS

THE FOLLOWING GENERAL CHECKS SHOULD BE PERFORMED THROUGHOUT THE EXTERIOR INSPECTION: CRACKS, DISTORTIONS, LOOSE FASTENERS; COVERS REMOVED; FUEL, OIL, HYDRAULIC LEAKS; ACCESS DOORS AND PANELS SECURELY FASTENED; AND EXTERNAL STORES SECURE.

NOTE

THE EXTERIOR INSPECTION BEGINS AT THE PILOT'S COCKPIT BOARDING LADDER AND PROGRESSES IN A CLOCKWISE DIRECTION AROUND THE AIRCRAFT. ITEMS THAT ARE ONLY ON ONE SIDE OF THE AIRCRAFT ARE SPECIFICALLY CALLED OUT AS LEFT SIDE (L) OR RIGHT SIDE (R). AFTER COMPLETION OF THE "AFT FUSELAGE AREA" CHECKS, REPEAT THE "CENTER FUSELAGE AND WING AREA" CHECKS FOR THE LEFT SIDE INSPECTION.

26512-1/41-0

Figure 3-2.

PREFLIGHT CHECK - CONTINUED

5. Nose gear and wheel well - CHECK
6. Pilot static head - CHECK (R)
7. Spikes and cones - EXTENDED
8. Weapons bay doors - CHECK
9. Rotating glove - CHECK
   Check for proper alignment and streamlined surface with slats retracted and proper deflection with slats extended.
10. Blow-in doors - CHECK

Ⓑ WINGS

1. Leading edges - CHECK
2. Trailing edge - CHECK
3. Wing seals - CHECK
4. Pylons and stores - CHECK

Ⓒ WHEEL WELL

Check for security and fluid leaks, pneumatic pressures, hydraulic reservoir level.

CONTINUED ON NEXT PAGE

PREFLIGHT CHECK - CONTINUED

Ⓓ TAIL SECTION

1. Ventrals - CHECK
   Check general condition.

2. Engine bay doors - CHECK

3. Tail skag - CHECK

4. Arresting hook - CHECK

5. Engine exhaust nozzles/ejectors - CHECK

6. Control surfaces - CHECK

7. Cooling turbine inlet - CHECK

8. Ram air outlet - CHECK

Ⓔ PYLON PANELS

1. Panel doors - SECURE

NORMAL COCKPIT ENTRY

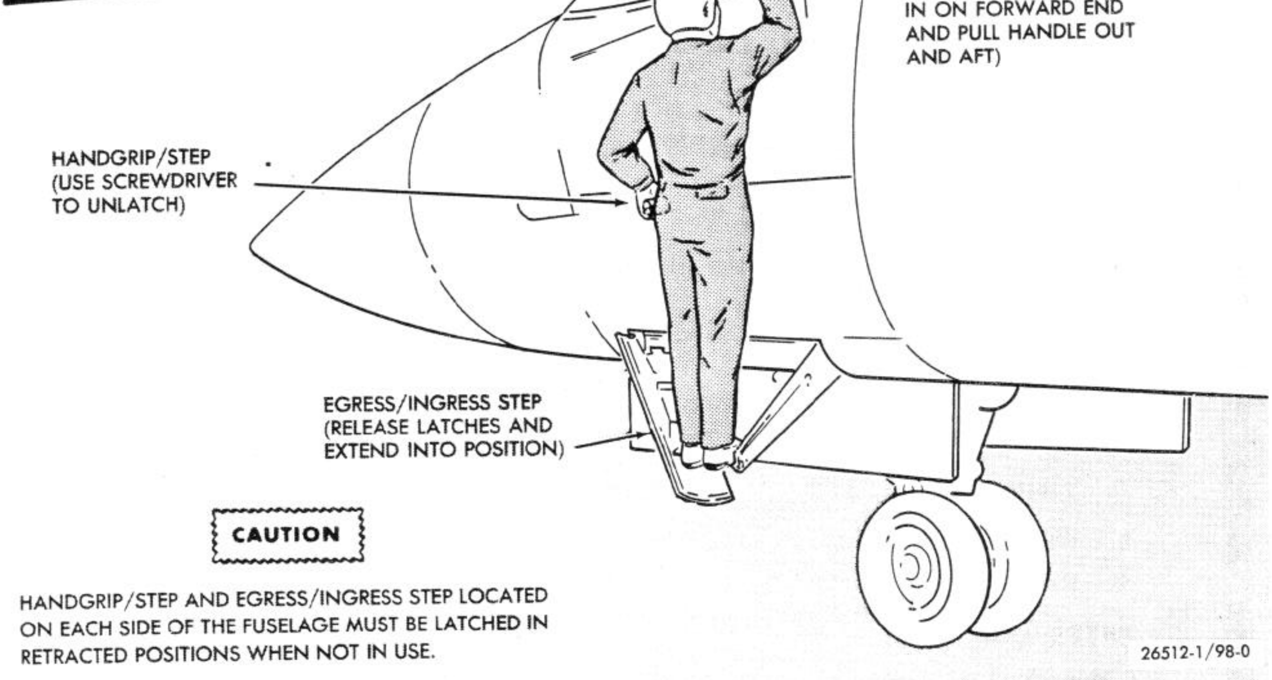


Figure 3-3.

PRESTART PROCEDURES

EXTERNAL POWER - OFF

1. Ejection handle safety pins (2) - INSTALLED
   Before entering the cockpit, check that a safety pin is installed in each ejection handle.
2. Severance and flotation, recovery parachute release and auxiliary flotation handle safety pins (2) - REMOVED
   Upon entering the cockpit, check that the above safety pins are removed.
3. Emergency pressurization bottle pressure - 3000 psi minimum at 70°F.
4. Emergency oxygen bottle pressure - 1800 psi minimum at 70°F.
5. Survival equipment - CHECK
   Check survival equipment for security.
6. Upper and lower torso restraint harness - CHECK
   Check the condition of the restraint harness.
7. Inertia reel - CHECK
   Check operation of the inertia reel in the locked and unlocked position.
8. Oxygen, communications and personnel services - ATTACH
9. Oxygen flow - CHECK
   Check that there is a normal flow of oxygen.
10. Landing gear handle - DOWN
11. Auxiliary brake handle - PULL
    Check brake cycles remaining gage.
12. Selective stores jettison switch - OFF
13. Ordnance switches - SAFE
14. Electrical test switch - OFF
15. HSI command switch - TACAN
16. LDG/TAXI LT switch - OFF
17. Hook-by-pass switch - FIELD
18. Spoiler brake/DLC switch - OFF
19. Spike switches (2) - NORM
20. Augmented Wave-Off (AWO) switch - OFF
21. Compass mode and latitude correction - SLAVED and SET
22. Interphone panel - SET
23. AFCS disconnect switch - NORM
24. Control system switch - AUTO
25. Auxiliary pitch trim switch - STICK
26. Slats/flaps switch - NORM

CONTINUED ON NEXT PAGE

PRESTART PROCEDURES - CONTINUED

27. Rudder authority switch - AUTO
28. Automatic carrier landing (ACL) switch - OFF
29. Damper switches (3) - DAMPER
30. Launch bar switch - UP
31. Slats/flaps handle - CORRESPONDING
32. Throttles - OFF
33. Throttle lock - DISENGAGE
34. External lights master switch - AS DESIRED
35. Speed brake switch - OFF
36. Engine start switch - OFF
37. APCS air temperature switch - CORRESPONDING
38. APCS switch - OFF
39. Throttle friction lever - FULL DECREASE
40. Wing sweep handle - CORRESPONDING
41. Flight instrument reference select switch - PRIMARY
42. Aerial refuel switch - RETRACT
43. Fuel tank pressurization switch - AUTO
44. Fuel dump switch - OFF
45. Fuel transfer selector - OFF
46. Engine fuel feed selector - OFF
47. Rain removal switch - OFF
48. Probe heaters switch - OFF
49. Engine anti-icing switch - AUTO
50. Anti-skid switch - ON
51. Utility hydraulic system isolation switch - ON
52. Cabin air distribution lever - AFT
53. Lighting panel - SET AS DESIRED
54. Malfunction and indication lights dimmer switch - AS DESIRED
55. Circuit breakers - IN
56. Publications and flight data - CHECK
    Check that all applicable current flight information publications are aboard.

CONTINUED ON NEXT PAGE

PRESTART PROCEDURES - CONTINUED

57. Ground check panel - SET

    a. Fire warning - NORM

    b. CADC switch - POWER

    c. Ground ignition switch - NORM

    d. Gyro switch - GYROS

    e. Computer switches (3) - OFF

CAUTION

Do not apply electrical power to the flight control computers unless air conditioning is available. Without cooling air the computers will overheat after 3 to 5 minutes operation with power on. For starts without ground air conditioning leave computer switches (3) to OFF position until after engine start. This caution also applies to the UHF, IFF and TACAN controls, radar altimeter and inertial navigation platform.

58. Cabin pressurization selector switch - NORM

59. Air conditioning mode selector switch - AUTO

60. Cockpit temperature selector - 10 O'CLOCK

61. Air source selector - BOTH

62. Emergency generator switch - AUTO

63. Generator switches (2) - ON

64. IFF/SIF - OFF

65. UHF #2 - OFF

66. TACAN - OFF

67. IFF antenna selector switch - AUTO

68. Alternate gear down handle - IN

69. Communications antenna selector switch - AUTO

70. UHF #1 - OFF

71. Hook switch - UP

72. Emergency arresting hook handle - IN

73. 12th stage bleed switches (2) - CLOSE

74. 6th stage bleed switches (2) - AUTO

75. Fire pull handles - IN

76. TACAN/ADF switch - TACAN

77. Altimeter - SET TO FIELD ELEVATION

78. Navigation computer - OFF

CONTINUED ON NEXT PAGE

PRESTART PROCEDURES - CONTINUED

EXTERNAL POWER - ON

1. External power switch - ON
   Check that the electrical power flow indicator displays TIE.

2. Landing gear indicators - DOWN

3. Speed brake indicator - UP

4. Interphone panel - CHECK
   Pull mixer knobs to ON and adjust volume on those functions that are to be used.

5. Seat and rudder panels - ADJUST

6. Malfunction and indication lamps test - CHECK
   Check for illumination of all indicator lamps.

7. Oil quantity - CHECK and TEST
   Check that indicators show 8 to 16 quarts. Depress and the oil quantity indicator test button and check that the test readings are 5 qts left and 5.7 qts right. Then, release test button and check that indicators return to original readings.

8. Oxygen quantity - CHECK and TEST
   Check that oxygen quantity is adequate for mission. Depress oxygen quantity test button; and indicator should decrease to zero. Note that the OXY LOW caution lamp illuminates when indication is less than 2 liters. Release test button and note that the OXY LOW caution lamp goes out and that the indicator returns to the original value.

9. Fuel quantity - CHECK and TEST
   Check fuel quantity in all tanks plus totalizer. Momentarily depress the fuel quantity indicator test button and check that all indicator readings decrease. Release test button and check that indicators return to original readings.

10. Fire detect circuit - TEST
    Check that the left and right fire warning lamps illuminate while holding the switch in the TEST position.

11. Engine fuel feed selector - FWD then AUTO
    Place the selector to FWD and check that fuel pump low pressure indicator lamps number 1-4 blink and stay out. Place the selector in AUTO and check that fuel pump low pressure indicator lamps number 5 and 6 blink and stay out.

12. Fuel transfer selector - WING
    Check that fuel pump low pressure indicator lamps 7 through 10 blink and go out if fuselage fuel tanks are full and wings are full.

BEFORE STARTING ENGINES

1. Wheels - CHOCKED
2. Fire bottle - MANNED
3. Engine inlet and exhaust area - CLEAR

WARNING

Suction in proximity to engine inlets is sufficient to severely injure personnel or draw foreign objects into the engine with resultant engine damage. Danger areas aft of the aircraft are created by high exhaust temperatures and velocities. Ear protection devices should be worn by ground personnel due to high engine noise levels.

Note

Whenever practicable, start the engines with the aircraft oriented so as to avoid tailwind components which cause higher than normal temperatures during engine start.

STARTING ENGINES

The engines cannot be started simultaneously; however, either engine can be started first. This procedure establishes starting the right engine (No. 2) first. Check that ground air conditioning is disconnected, engine starter air available and plane captain signals clearance for start.

1. Engine start switch - RIGHT ENGINE
2. Right throttle - raise to CRANK position
   a. Engine rpm increases
   b. Oil pressure - check for positive indication
   c. R PRI and UTIL HYD caution lamps out by 16.5% rpm and before 1100 psi
3. At 16.5% rpm - right throttle to IDLE
   a. Fuel flow - 1100 pph maximum. If fuel flow exceeds 1100 pph during acceleration to idle rpm, shut down the engine to prevent a hot start (705°C maximum TIT).
   b. Lightoff should occur within 7 seconds after placing throttle to IDLE. If a wet start occurs, place throttle in CRANK and clear the engine.
   c. R GEN caution light out by 50% rpm
   d. Idle rpm - 54% to 62%. In event of a hung start (rpm does not accelerate to idle), retard throttle to OFF.
   e. Idle fuel flow - 500 pph minimum
1100 pph maximum
   f. Nozzle position - OPEN (10)
   g. Idle oil pressure - 30 to 50 psi
   h. Hydraulic pressure indicators (2) - 2950 to 3250 psi

CONTINUED ON NEXT PAGE

STARTING ENGINES - CONTINUED

4. Electrical power flow indicator - TIE

5. External power switch - OFF

Signal Plane Captain To Disconnect External Power and Engine Starter Air.

Plane Captain Signals Disconnect Complete, Ready for Crossbleed Start.

6. Right engine to 75% rpm for crossbleed start. Retard throttle to IDLE when left engine reaches 40% rpm.

Start Left Engine By Repeating Steps 1 Through 3 Substituting Left for Right.

7. Engine start switch - OFF

8. Electrical power flow indicator - NORM

AFTER STARTING ENGINES

1. Emergency generator test - ON, TEST then AUTO

   ON - Emergency generator indicator lamp illuminates after 3 seconds. Power flow indicator should display NORM.

   TEST - Emergency generator indicator lamp stays illuminated, and the power flow indicator should display a crosshatch. Nozzle position indicators should remain steady.

   AUTO- Indicator lamp goes out and power flow indicator displays NORM. Nozzle position indicators indicate open.

2. Communications, navigation, identification (CNI) equipment - ON

CAUTION

Ensure air conditioning is available prior to operation of the electronic equipment.

3. Radar altimeter - ON

4. Flight control computers (3) - ON

5. Landing gear ground safety locks - REMOVED
   Check visually with ground crewman.

6. Crew module ejection handle safety pins (2) - REMOVED and STOWED

7. Boarding ladders - STOWED

8. Engine/generator cooling air - Plane captain check for ejector airflow

9. G-valve test button - DEPRESS
   Momentarily depress the g-valve test button to check for instant g-suit pressurization.

10. Suit vent controls - AS DESIRED

11. Caution/warning lamps - OUT
    Except PROBE HTRS and CANOPY

BEFORE TAXIING PROCEDURES

It is assumed that the before taxi checks are performed with the wings initially in the full swept condition.

1. Flight controls - CLEAR
   Check with plane captain.

2. Flight control system - CHECK

   a. Damper switches (3) - OFF
      Check illumination of damper caution lamps (3).

   b. Damper switches (3) - DAMPER

   c. AFCS reset button - DEPRESS
      Check that all damper/channel/gain changer caution lamps are out.

   d. Controls - CYCLE
      Check for freedom and direction of movement and that no damper or channel caution lamps illuminate.

   e. Spoiler throw - CHECK
      Depress master test button and check spoiler deflection with lateral stick deflection.

   f. Symmetric spoiler lock-out - TEST
      With master test button depressed, momentarily hold spoiler test switch to INBOARD then OUTBOARD position and check that lock-out circuit causes selected spoiler set to lock in the DN position and spoiler caution lamp to illuminate. Deflect lateral control and insure that other set of spoilers is operative. Reset lock-out circuitry by depressing spoiler reset button.

   g. Direct Lift Control (DLC) - TEST
      With master test button depressed, place the DLC switch in ENGAGE position; note DLC/BRAKE advisory lamp go out and inboard spoilers raise. Move DLC control button on stick and check position indication and fluctuation in PRIMARY HYD pressure. Check that DLC switch returns to STBY and advisory lamp illuminates when either master test button is released or speed brake switch is placed to IN.

   h. Spoiler brake - TEST
      With spoiler brake switch in BRAKE, check deflection of all spoilers and illumination of DLC/BRAKE advisory lamp with master test button depressed. Check spoilers DN with either throttle 3 degrees out of IDLE position.

   i. Null trim button - DEPRESS
      Check nulling of all surfaces and null trim lamp illumination.

   j. Stability Augmentation System (SAS) - TEST

      (1) Depress master test button and switch to SURFACE MOTION - surfaces deflect aircraft nose down, left wing down, and left rudder. Control system caution lamps should not illuminate.

      (2) Depress master test button and switch to SURFACE MOTION & LIGHTS - Surfaces deflect aircraft nose down, left wing down, and right rudder. Check damper (3) and channel (3) caution lamps and pitch and roll gain changer (2) caution lamps illuminate.

      (3) AFCS reset - DEPRESS
          Check all flight control caution lamps out.

   k. LSTC - TEST

   l. AYC - TEST

CONTINUED ON NEXT PAGE

BEFORE TAXIING PROCEDURES - CONTINUED

3. Rudder authority - TEST

   a. Rudder authority AUTO and control system switch in TO&L - check full rudder throw.

   b. Rudder authority FULL and control system switch in AUTO - check full rudder throw and illumination of RUDDER AUTHORITY caution lamp.

   c. Rudder authority AUTO and control system switch in AUTO - check restricted rudder throw and caution lamp out.

4. Autopilot - TEST

   a. Pitch and roll damper switches - AUTOPILOT

   b. Control stick steering - CHECK
   Move control stick and check that REF NOT ENGAGE lamp illuminates. Lamp will go out when stick is returned to neutral.

   c. ALTITUDE HOLD and CONSTANT TRACK switches - ENGAGED
   REF NOT ENGAGED lamp illuminates.

   d. Reference engage switch - ENGAGE
   REF NOT ENGAGED lamp goes out.

   e. Move control stick and release
   REF NOT ENGAGED lamp illuminates.

   f. Reference engage switch - ENGAGE
   REF NOT ENGAGED lamp goes out.

   g. Autopilot emergency release lever - DEPRESS
   Check that pitch and roll damper switches return to the DAMPER position and ALT HOLD and CONSTANT TRACK switches return to OFF.

5. AFCS override switch - DISC ORIDE
   Check deactivation of controlling functions.

6. Trim - CHECK

   a. Pitch series trim - CHECK
   Turn pitch damper OFF and check nose UP/DOWN control and return to STICK.

   b. Rudder trim - CHECK
   Check operation to left and right.

   c. Pitch parallel trim - CHECK
   Check operation in both directions.

   d. Roll trim - CHECK
   Check operation in both directions.

   e. Null trim button - DEPRESS
   Check nulling of all surfaces and illumination of null trim lamp.

7. Launch bar - CHECK and UP
   Place switch to DOWN and observe indication. Return switch to UP and check for launch bar stowed indication.

CONTINUED ON NEXT PAGE

BEFORE TAXIING PROCEDURES - CONTINUED

8. Spikes (2) - TEST
   Depress L/R test button and observe SPIKE caution lamp illuminates during spike transition to full aft and expanded position. Release test button and observe same transient indication. Plane captain verify spike position.

   a. Ground check panel - CLOSE

   b. Nose wheel steering - CHECK

   c. Launch bar - CHECK and RETURN TO UP

9. APCS - TEST
   Plane captain position ADD probe to full clockwise position. Switch to STBY and note illumination of APCS advisory lamps. With master test button depressed switch to ENGAGE, note APCS advisory lamp goes out; throttles should drive to minimum rpm for APCS command. Check manual override forces for disengagement of APCS together with speed brake IN and release of master test button electrical disengagement features. Return APCS switch to OFF.

10. Hook-by-pass - TEST
    Momentarily hold in TEST position and check illumination of indexer and approach lamps.

11. Hook operation - TEST
    Place arresting hook switch in DOWN position and check illumination of HOOK warning lamp during hook lowering; lamp should go out with hook resting on the deck. Deflect rudder pedals and check activation of nose wheel steering with hook down and weight-on-wheels. Place hook switch to UP and again check for transient illumination of HOOK warning lamp during retract cycle.

12. Aerial refueling probe operation - TEST
    Place switch to EXTEND and check for full extension. Return switch to RETRACT and note operation.

13. 12th stage bleed - TEST
    Note individual engine rpm droop with opening of bleed air valves. Return bleed to CLOSED position.

14. 6th stage bleed - TEST
    Note individual engine rpm droop with opening of bleed air ports. Return bleed to AUTO position.

15. Engine inlet anti-icing - TEST
    Place switch to MAN position and note change in engine rpm. Return switch to AUTO.

16. Probe heaters - CHECK
    Place switch in PRI and note that PROBE HTR caution lamp goes out after a time delay. Place switch to SEC and note that lamp goes out. Return switch to PRIMARY.

17. Rain removal - CHECK
    Place switch to RAIN REMOVAL and check for hot bleed air on left forward windshield. Return switch to OFF.

18. Inertial navigation platform - ALIGN
    Check PRI ATT/HDG caution lamp out.

19. Instrument reference select - CHECK
    Place switch to STBY and note attitude/heading disparity with platform reference. Return switch to PRI. AFRS needle synchronized.

20. Radar altimeter - TEST

CONTINUED ON NEXT PAGE

BEFORE TAXIING PROCEDURES - CONTINUED

21. Nose wheel steering - CHECK

Note

Depending on the operational situation, the wings may be swept forward to 16° and slats/flaps extended fully at this time prior to leaving the flight line.

22. Configuration check by plane captain:

    Wing Sweep
    Slats/Flaps
    Spoiler Brakes
    Spikes
    Weapons Bay Doors
    Access Panels
    Gear Down Lock Pins (2)
    Armament Safety Pins

TAXIING PROCEDURES

1. Chocks - REMOVE
2. Parking brake - RELEASE while holding brakes
3. Flight instruments - CHECK
   Record altimeter instrument error.
4. Control system switch - AUTO
5. 6th stage bleed - AUTO
6. 12th stage bleed - CLOSE
7. Wing sweep - 16°
8. Slats/flaps - FULL DOWN (40°)
9. Control throw - CHECK
10. Null trim - SET
11. Fuel feed and quantity - CHECK

WARNING

The canopy latch handle lock tab shall be left unlocked (not flush) for takeoff, to insure access to cockpit, if needed, for crew rescue on the ground.

12. Canopies (2) - CLOSE and LOCK
13. Oxygen - ON
14. Warning/caution lamps - OUT

CONTINUED ON NEXT PAGE

TAXIING PROCEDURES - CONTINUED

15. Harness - LOCK

16. Crew module pins - REMOVE

WARNING

Leave canopy external emergency release initiator safety pins installed throughout the flight to prevent inadvertent actuation of the canopy emergency release system because of failure of the external release mechanism.

TAKEOFF PROCEDURES

Prior to taking the duty runway, the takeoff check list shall be completed, using command response on the ICS. Takeoffs will be made with ICS HOT MIKE selected.

WING
TRIM
SLATS/FLAPS
FUEL
CONTROLS
BRAKES
PINS
HARNESS
WARNING LTS

1. Wings - 16° SWEEP

2. Slats/flaps - FULL DOWN (40°)

3. Controls - FREE

4. Trim

   a. Null trim lamp - ON

   b. Check Rudder - 0°

   c. Check Horizontal tails - 0°

5. Fuel quantity and distribution - CHECK

6. Engine fuel feed selector - AUTO

7. Pins - REMOVED

8. Harness - LOCKED

9. Warning/caution lamps - OUT

10. MCO takeoff report - MCO READY FOR TAKE OFF

AFTER TAKEOFF - CLIMB

When the aircraft is definitely airborne and a positive rate of climb has been established:

1. Landing gear handle - UP
   Check that the landing gear and speed brake indicate UP before exceeding 285 KIAS.

   Note

   Inability to raise landing gear handle when airborne without depressing landing gear handle lock release button, is indicative of a malfunction of landing gear weight on wheels switch or launch bar uplock switch.

2. Slats/flaps handle - UP
   Do not initiate slats/flaps retraction at angles-of-attack exceeding ___ units. Check that the slats/flaps indicate UP before exceeding limit speed of 285 KIAS.

   Note

   For heavy gross weight takeoffs or high drag loadings retract the flaps to 15° initially and fully retract slats/flaps by the limit airspeed.

3. Wing sweep handle - 26°
   (Prior to 300 KIAS)

4. Utility hydraulic system isolation switch - ISOL
   After the landing gear and the slats/flaps have been fully retracted, place the isolation switch in the ISOL position.

5. Spoiler brake switch - OFF

6. Fuel quantity and distribution, feed and transfer - CHECK

7. Engine instruments - CHECK

8. Accelerate to the recommended climb speed as indicated in Section XI, Performance Data.

9. Canopy latch handle lock tab - FLUSH
   Depress the canopy latch handle against the CLOSED position and snap the spring loaded latch handle lock tab into the LOCKED (flush) position.

CRUISE

Cruise control data for various loadings are contained in Section XI, Performance Data. Refer to Section IV, Flight Procedures, for information pertaining to the flight characteristics of the aircraft.

DESCENT

1. Cabin air distribution control lever - FWD (DEFOG)

2. Fuel quantity and fuel panel - CHECK

CONTINUED ON NEXT PAGE

DESCENT - CONTINUED

3. Hydraulic pressure - CHECK
   Check for 2950 to 3250 psi indication and pressure recovery after control displacement.

4. Spike switches (2) - NORM

5. 6th stage bleed switches (2) - AUTO

6. Wing sweep - 26°

7. Altimeter - RESET

BEFORE LANDING

1. Wing sweep - 26°

WARNING

Do not enter the traffic pattern with wing sweep greater than 26 degrees. The wings must be at 26 degrees or less to allow slats/flaps extension and to prevent excessive sink rates. Recovery from a high sink rate may be impossible at traffic pattern altitudes.

2. Armament - SAFE

3. Anti-skid switch - ON

4. Fuel quantity and feed - CHECK

5. Hook - UP

6. Canopy latch handle lock tab - UNLOCK

WARNING

Unlock the canopy latch lock tab prior to landing, to insure access to cockpit, if needed, for crew rescue on the ground.

LANDING

The landing check list should be completed before the final approach phase using command response on the ICS. Landings will be made with ICS HOT MIKE selected. For a normal landing, fly the pattern as illustrated in figure 3-4.

1. Wing sweep - 16° (Below 300 KIAS)

2. Landing gear handle - DOWN
   Extend the landing gear after the airspeed is below 285 KIAS. Check that the warning lamp in the gear handle is out, the landing gear (including tail bumper) indicate DOWN, and the speed brake indicates UP. Extension of the landing gear deactivates the speed brake mode and the speed brake switch can remain in the HOLD position to enable APCS and DLC engagement.

CONTINUED ON NEXT PAGE

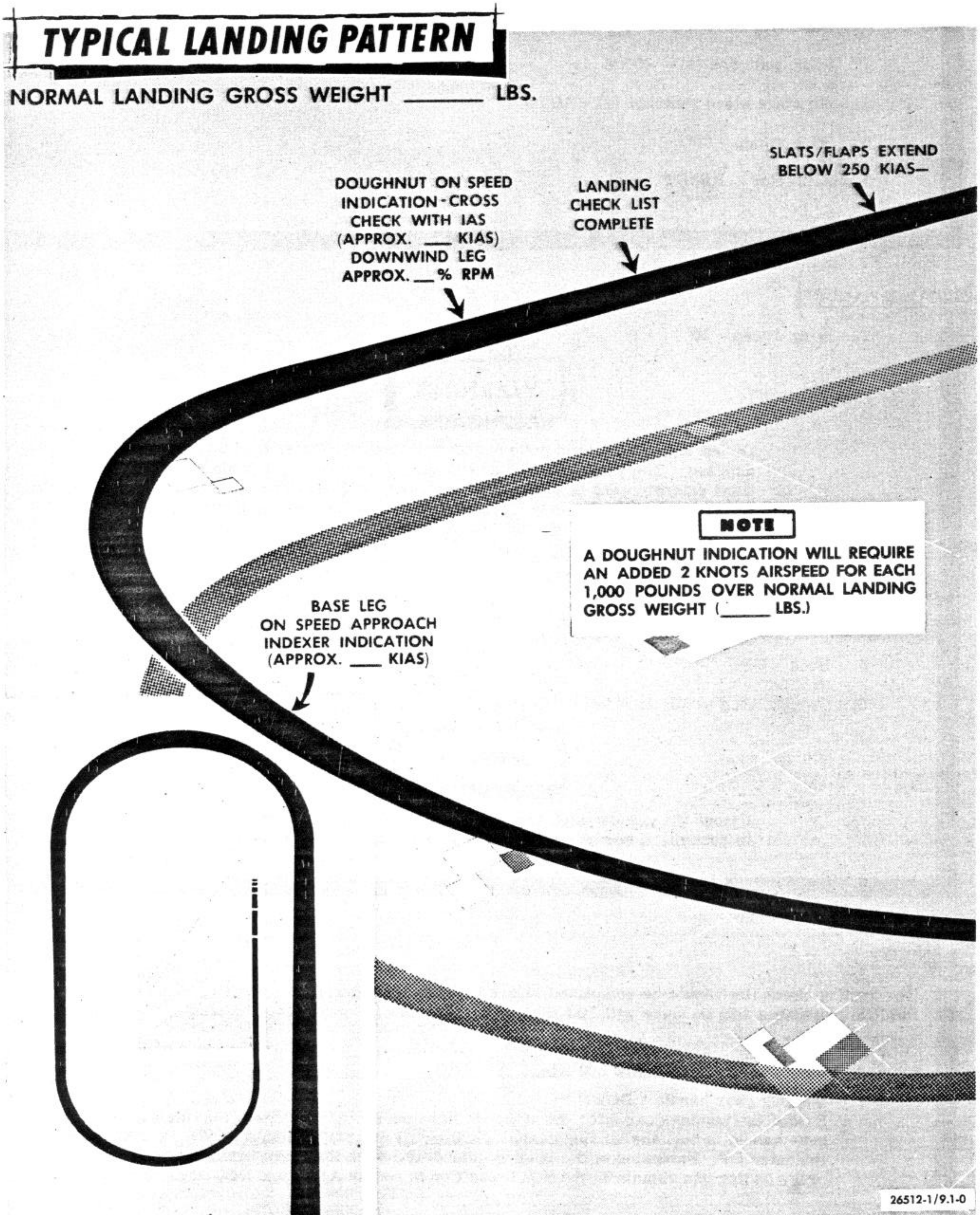


Figure 3-4. (Sheet 1)

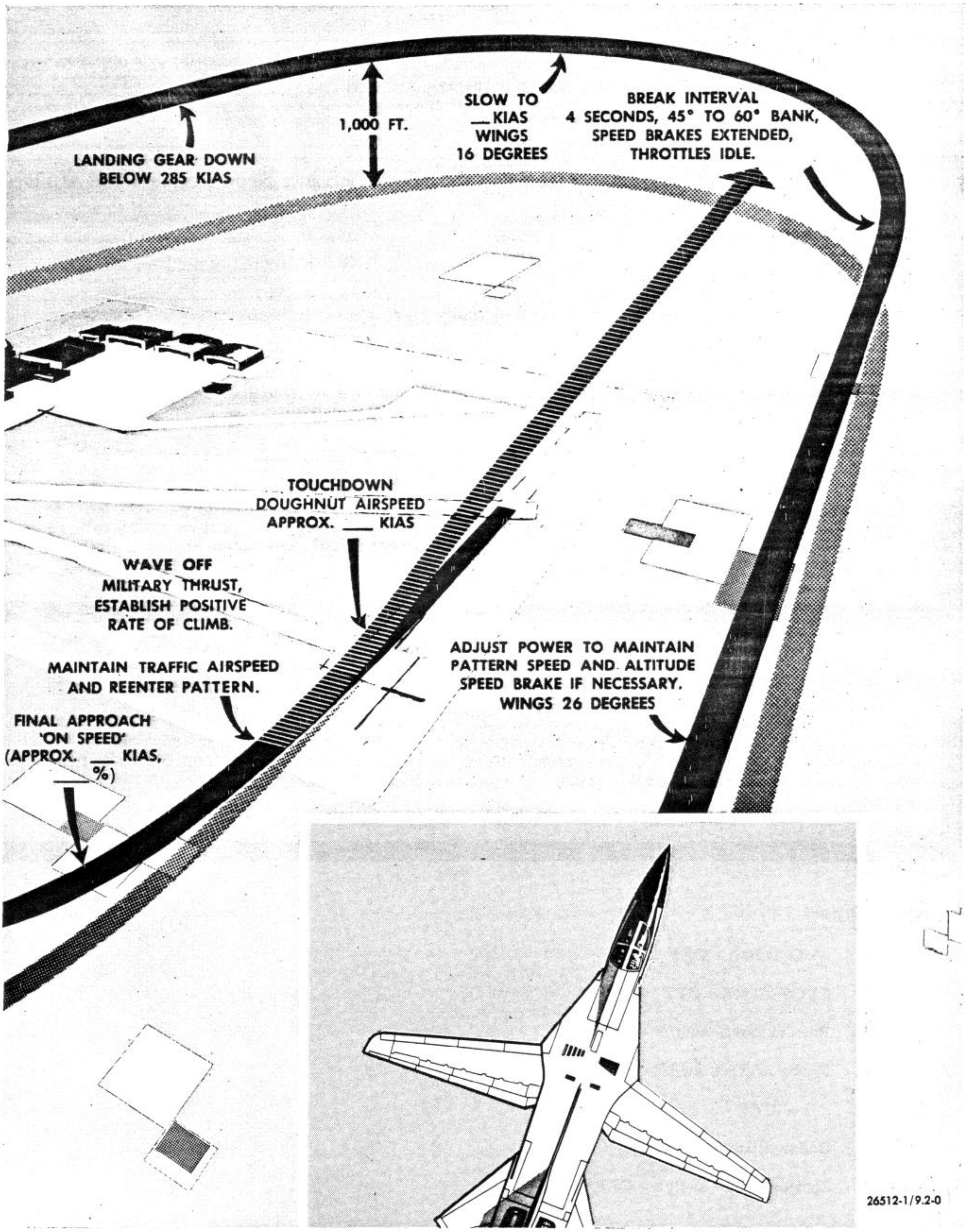


Figure 3-4. (Sheet 2)

LANDING - CONTINUED

3. Slats/flaps handle - DOWN
Select slats/flaps down after the airspeed is below 250 KIAS. Check position indicators to assure that the high-lift devices have moved to the selected position.

4. Throttle friction - FULL DECREASE

5. AWO switch - AWO for single engine approaches.
To activate AWO, place the switch to the AWO position and advance the throttle to MIN A/B to effect a light-off.

6. APCS - ENGAGE

7. DLC - ENGAGE

8. Brakes - CHECK pedals and cycles remaining gage.

9. Harness - LOCK

WAVE-OFF

The decision to wave-off should be made as early as possible. When the decision to wave-off is made, smoothly advance the throttles to MIL or MAX A/B, as desired, to establish a positive acceleration and climb. Rotate the aircraft to an angle-of-attack not exceeding___units to stop the sink rate while simultaneously placing the speed brake switch to RETRACT (deactivates DLC and APCS).

TOUCH-AND-GO LANDING

The landing check and pattern procedures performed for touch-and-go and final landings are the same. After touchdown, advance the throttles to MIL or MAX A/B while checking engine instruments and effect a normal takeoff. If remaining in the landing pattern, leave the landing gear and flaps DOWN and engage APCS and DLC when established downwind. Comply with the After Takeoff - Climb check list if departing the landing pattern.

AFTER LANDING

1. AWO switch - OFF
2. APCS switch - OFF
3. Rain removal switch - OFF
4. Probe heaters switch - OFF
5. IFF - OFF
6. Radar altimeter - OFF
7. Spoiler brake switch - OFF

CONTINUED ON NEXT PAGE

AFTER LANDING - CONTINUED

8. Slats/flaps handle - UP
   Check position indicators to assure retraction.

9. Wing sweep - 72.5° or AFT LIMIT
   Depending on store loading.

10. Null trim button - DEPRESS
    Check for illumination of null trim lamp and nulling of surface positions.

11. Boarding ladders (L/R) - EXTEND

ENGINE SHUTDOWN

1. Wheels - CHOCKED

   CAUTION

   To prevent damage to brakes from overheating, do not set the parking brake until brake discs have had time to cool.

2. Landing gear ground safety locks - INSTALL

3. Armament safety pins - INSTALL

4. CNI equipment - OFF

5. Engine fuel feed selector - OFF

6. External electrical power switch - AS DESIRED
   Place switch to ON position after external power cables have been connected to the aircraft.

7. Oxygen - OFF

8. SAS computer switches (3) - OFF

9. Air conditioning mode switch - OFF

10. Oil quantity - CHECK
    Run up engines to 70% rpm for 30 seconds prior to shutdown to scavange oil.

11. Right engine throttle - OFF
    Check hydraulic pressure and electrical power flow indicator for TIE.

12. Left engine throttle - OFF
    Without external electrical power, check that the emergency generator comes on the line when the generators disconnect from the buses while the hydraulic pressure bleeds off.

BEFORE LEAVING AIRCRAFT

1. All switches and controls - OFF, NORMAL, or SAFE

2. Crew module ejection handle safety pins - INSTALL

3. Crew module severance and flotation and recovery parachute release, and auxiliary flotation handle safety pins - INSTALL

part 4
CARRIER-BASED PROCEDURES (PILOT)

TO BE SUPPLIED AT A LATER DATE

part 5

SHORE-BASED PROCEDURES (MCO)

TABLE OF CONTENTS

PREPARATION FOR FLIGHT

Accomplishing the pre-flight inspection of the airborne missile control system and the inertial navigation system will be the prime responsibility of the missile control officer prior to each mission. This inspection will include a review of ground crew and armament personnel actions regarding the weapons system, inspection of all weapon installations, and operational check of the integrated attack-navigation equipment. A thorough crew briefing will be conducted prior to accomplishing this inspection. This briefing will familiarize the crew with the type of mission being conducted, emergency procedures, crew coordination, and individual responsibility and cooperation through the planned flight.

Note

- All procedures contained in the following pages pertain primarily to the responsibilities of the MCO. Cockpit preflight inspection should be conducted by the pilot and the MCO simultaneously to ensure that all steps are successfully completed, and to satisfy individual crew requirements.

- The pilot's takeoff and landing checks shall be completed using command response on the ICS. Takeoffs and landings will be made with pilot's and MCO's ICS HOT MIKE selected.

PRESTART PROCEDURES

EXTERNAL POWER - OFF

1. Ejection handle safety pins (2) - INSTALL
   Before entering the cockpit, ensure that a safety pin is installed in each ejection handle.

2. Emergency cabin pressurization bottle pressure - 3000 psi at 70°F

3. Emergency oxygen bottle pressure - 1800 psi

4. Upper and lower torso restraint harness - CHECK
   Check the condition of the restraint harness.

5. Inertia reel - CHECK
   Check operation of the inertia reel in the locked and unlocked position.

6. Oxygen, communications, and personal services - ATTACH

7. Oxygen flow - CHECK
   Check for normal flow of oxygen

8. NAV mode selector switch - OFF

CONTINUED ON NEXT PAGE

PRESTART PROCEDURES - CONTINUED

9. Platform align switch - SET
NORM for gyrocompass alignment; RAPID ALIGN if heading was previously stored.

Note

If the aircraft has been moved since the heading was stored, a gyrocompass alignment must be accomplished.

10. Position control selector - MAN FIX

11. MAG VAR - SET

12. UHF #2 - OFF

13. Circuit breakers - IN

EXTERNAL POWER - ON

1. External power switch - ON
Check electrical power flow indicator displays TIE.

2. Interior lighting control panel - SET AS DESIRED

3. Interphone panel - CHECK
Pull appropriate mixer knobs to ON and adjust volume.

4. Oxygen quantity - CHECK and TEST
Check oxygen quantity is adequate. Depress oxygen quantity test button, indicator shall decrease to zero. When indicator is less than 2 liters OXY LOW caution light shall illuminate. Release test button and note OXY LOW light goes out and indicator returns to original value.

5. UHF #2 - ON and CHECK

INERTIAL PLATFORM ALIGNMENT

GROUND ALIGNMENT

If ground power and ground air conditioning are not available, wait until aircraft power and air conditioning are available, then proceed as follows:

CAUTION

Do not apply electrical power to the inertial navigation system unless air conditioning is available. Damage due to overheating will occur after 3 to 5 minutes operation without air cooling.

1. Platform alignment switch - NORM

2. Mode selector switch - HEAT
The HEAT lamp will illuminate indicating system warm-up.

3. Position control selector - MAN FIX
Set local latitude and longitude coordinates.

4. Position control selector - PRES POS
Check latitude and longitude inserted in the computer

CONTINUED ON NEXT PAGE

INERTIAL PLATFORM ALIGNMENT - CONTINUED

5. MAG VAR - SET
   Set local magnetic variation.

When The HEAT Lamp Goes Out:

At ambient temperatures above 40°F the HEAT lamp will go out in approximately 2 to 3 minutes.

6. Mode selector switch - ALIGN
   The INEG STOP lamp will illuminate and the ERROR lamp may momentarily flicker.

CAUTION

If the ERROR lamp remains illuminated, turn the mode selector switch to OFF.

7. ALIGN lamp - MONITOR
   Approximately 90 seconds after selecting ALIGN, the INEG STOP lamp will go out and the ALIGN lamp will illuminate steady.

When The ALIGN Lamp Flashes:

Approximately 5 to 10 minutes after selecting ALIGN, the ALIGN lamp flashes indicating alignment is completed.

Note

Monitor magnetic heading synchronization indicator and allow meter to achieve a fine null if possible before selecting a navigation mode. The quality of alignment is proportional to the quality of the null.

8. Mode selector switch - GC or SR NAV
   Select the appropriate operating navigation mode for the mission to be flown.

9. Destinations - STORED and SET

Note

If the aircraft is to be flown at a later time, the aircraft true heading can be stored in the navigation computer by selecting RAPID ALIGN before turning the mode selector switch to OFF.

RAPID ALIGNMENT

Rapid alignment shall be accomplished under shore-based conditions only.

1. Perform ground alignment as outlined under ground alignment procedures.

2. Platform alignment switch - RAPID ALIGN

3. Mode selector switch - OFF

When The Aircraft Is Ready For Pre-Flight and Providing The Aircraft Has Not Been Moved Since Ground Alignment:

4. Mode selector switch - ALIGN

5. ALIGN lamp - MONITOR
   During rapid alignment the ALIGN lamp will remain out until alignment is completed at which time the lamp will begin to flash.

CONTINUED ON NEXT PAGE

RAPID ALIGNMENT - CONTINUED

6. Platform alignment switch - NORM

CAUTION

If the ERROR lamp illuminates and an AUX navigation mode is selected with the platform control switch in RAPID ALIGN, the true heading shaft in the computer will lock. This condition will cause damage to the servo amplifier in the heading module.

7. Mode selector switch - GC or SR NAV
Select the appropriate operating navigation mode for the mission to be flown.

8. Destinations - STORED and SET.

BEFORE TAXIING PROCEDURES

If ground power and ground air conditioning are not available, wait until aircraft power and air conditioning are available, then proceed as follows:

1. Mode selector switch - ALIGN

2. Position control selector - PRES POS
Set local latitude and longitude

3. MAG VAR - SET
Set local magnetic variation. Monitor magnetic heading synchronization indicator. Allow the meter to achieve a fine null before selecting a navigation mode.

4. Platform indicator lamps - MONITOR
Monitor indicator lamps until ALIGN lamp begins to flash.

CAUTION

If the ERROR lamp remains illuminated, turn the mode selector switch to OFF.

5. Mode selector switch - GC or SR NAV
Select the appropriate operating navigation mode for the mission to be flown.

Note

- If alignment was accomplished on ground power a navigation mode must be selected prior to turning aircraft generators on. Changing power source will jeopardize alignment accuracy.

- Taxiing or towing the aircraft while the platform is being aligned will delay the alignment and jeopardize its accuracy.

6. Destinations - SET and STORED

7. G-valve test button - DEPRESS
Momentarily depress the g-valve test button, check for instant g-suit pressurization.

8. Suit vent controls - AS DESIRED

9. Taxi report to pilot - MCO READY TO TAXI

TAXIING PROCEDURES

1. Canopy - CLOSE and LOCK
2. Oxygen - ON
3. Harness - LOCK
4. Crew module pins - REMOVE
5. Navigation control panel - MONITOR
   a. Groundspeed - MONITOR
   b. RNG/CRS - CHECK
   c. VDIG/HSI - CROSS CHECK
6. Oxygen - ON
7. Warning/caution lamps - OUT
8. Harness - LOCK
9. Ejection handle pins - REMOVE

TAKEOFF PROCEDURES

Prior to taking the duty runway, the takeoff check list shall be completed using a command response on the ICS. Takeoff will be made with ICS HOT MIKE selected.

1. Wings - 16° Sweep
2. Slats/flaps - FULL DOWN (40°)
3. Controls - FREE
4. Trim
   a. Null trim - ON
   b. Check Rudder - 0°
   c. Check Horizontal tails - 0°
5. Fuel - quantity and distribution - CHECK
6. Engine fuel feed switch - AUTO
7. Pins - REMOVED
8. Harness - LOCKED
9. Warning/caution lamps - OUT
10. Takeoff report - MCO READY FOR TAKEOFF

IN FLIGHT OPERATION PROCEDURES

DESTINATION STORAGE

1. DDT switch - PRES POS
2. Position control selector - STORE INSERT
   When the position counters stop driving they will indicate aircraft coordinates.
3. DDT switch - DEST 1, 2, or 3
4. Latitude and longitude counters - SET
   Set after position counters stop driving.

CONTINUED ON NEXT PAGE

IN FLIGHT OPERATION PROCEDURES - CONTINUED

5. Position control selector - DEST POS
6. DDT switch - OFF or AS REQUIRED

MANUAL FIX UPDATING

Prior To Arrival Over The Check Point:

1. Position control selector - MAX FIX
   INTEG STOP lamp will illuminate.
2. Latitude and longitude counters - SET
   Slew coordinates of check point.

When Over Check Point:

3. FIX pushbutton - DEPRESS
   INTEG STOP lamp will go out.
4. Latitude and longitude counters - CHECK
   Check that latitude and longitude counters are driving.
5. Position control selector - PRESS POS
   Check counters for proper coordinates.

AUTOMATIC FIX UPDATING

Prior To Arrival Over Check Point:

1. Position control selector - AUTO FIX
2. Latitude and longitude counters - SET
   Slew coordinates of check point into destination channel of the computer

Note

If the checkpoint coordinates are stored in the computer, they can be inserted by selecting the appropriate DEST on the DDT selector.

When Over Check Point:

3. FIX pushbutton - DEPRESS

AUXILIARY NAV MODE

If platform malfunction occurs, the ERROR lamp will illuminate and the navigation system automatically enters the auxiliary mode.

1. Mode selector switch - AUX
   The ERROR lamp will remain illuminated until the AUX mode is selected.
2. Platform alignment switch - OFF
3. DDT switch - DISPLAY WIND
   Monitor wind values; change manually by slew controls, as required.
4. MAG VAR - MONITOR
   Periodically update from chart data as required.
5. Latitude and longitude counters - MONITOR
   Periodically update as required

LANDING PROCEDURES

The landing check list should be completed before the final approach phase using command response on the ICS. Landings should be made with ICS HOT MIKE selected.

1. Wing sweep - 16° (Below 300 KIAS)
2. Landing gear handle - DOWN
3. Slats/flaps handle - DOWN
4. Throttle friction - FULL DECREASE
5. AWO switch - AWO (for single-engine approach)
6. APCS - ENGAGE
7. DLC - ENGAGE
8. Brakes - CHECK, ANTI-SKID ON
9. Harness - LOCK

AFTER LANDING - IN CHOCKS

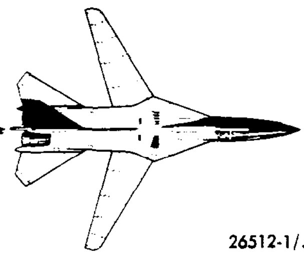
1. CNI equipment - OFF
2. Oxygen - OFF
3. NAV mode selector switch - OFF
4. UHF #2 - OFF
5. Crew module ejection handle safety pins - INSTALL

part 6

CARRIER-BASED PROCEDURES (MCO)

TO BE SUPPLIED AT A LATER DATE

section IV
FLIGHT PROCEDURES



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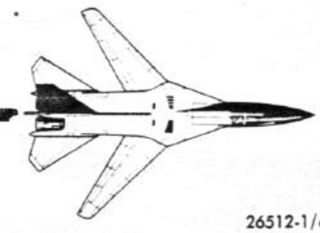
TO BE SUPPLIED AT A LATER DATE

section V
EMERGENCY PROCEDURES



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INTRODUCTION

Knowledge of the aircraft and emergency procedures must be reviewed on a regular basis to ensure that the crew will take the correct course of action when faced with difficulties. The initial training should be thorough in this respect. Above all, the crew must recognize and admit the emergency situation, then take positive steps in accordance with recommended procedures and good airmanship. Due to the many situations that can arise concerning emergencies, it is impossible to set an absolute policy.

The crew must weigh all the factors of a given situation and then take appropriate action for the particular situation. This manual discusses and preplans some likely courses of action and the recommended way of handling certain emergencies. The emergency procedure section should be referred to on a continuing basis. Reference to emergency procedures promulgated in NWP-41 is also required.

Note

As soon as possible, the pilot should notify the Missile Control Officer of the emergency and the intended action.

GROUND OPERATION EMERGENCIES

ENGINE FIRE OR OVERHEAT DURING START OR SHUTDOWN

1. Throttle . . . . OFF
2. Engine . . . . MOTOR

If Fire:

3. Fire pull handle . . . . PULL
4. All switches . . . . OFF
5. Abandon the aircraft.

EMERGENCY EGRESS

1. Upper torso harness buckle . . . . . RELEASE
2. Lower torso harness buckle . . . . . RELEASE
3. Hoses . . . . DISCONNECT
4. Abandon the aircraft.

TAKE-OFF EMERGENCIES

The take-off phase of flight is critical in that it affords the pilot a very short period of time in which to decide whether to continue or abort the takeoff. The pilot must have fixed firmly in mind the best course of action to be taken in any given situation. Prior to each flight the pilot should know:

1. Lift-off airspeed and take-off ground roll.
2. Refusal speed and distance, or line check speed and distance.
3. Single-engine performance (refer to Section XI).
4. Availability and location of arresting abort gear.
5. Surrounding terrain and obstructions.

ABORTED TAKEOFF

1. Throttles . . . . IDLE (OFF FOR FIRE)
2. Wheel brakes. . . . . APPLIED
3. Hook . . . . DOWN
4. Spoiler brakes . . . . BRAKE

ENGINE FAILURE/TAKEOFF CONTINUED

The pilot's reaction and his ability to maintain directional control, altitude, or climb depends upon the gross weight of the aircraft, air density, ambient temperature, and the thrust of the good engine. Prior to each flight it is essential that the pilot know the single-engine climb speed for his aircraft configuration. If this speed has not been attained, the following steps are considered optimum to continue safe flight:

1. Operating engine . . . . MAX. THRUST
2. External stores . . . . JETTISON (AFTER AIRBORNE)
3. Fuel. . . . . DUMP
4. Establish single-engine climb.
5. Landing gear . . . . . UP (AFTER AIRBORNE)
6. Fly straight ahead. Attempt no turns until safe airspeed and altitude are attained (terrain permitting).
7. Flaps. . . . . UP (AFTER SAFE AIRSPEED IS ATTAINED)
8. Failure other than mechanical. . . . . . ATTEMPT AN AIRSTART

For Obvious Mechanical Failure:

9. Throttle (failed engine) . . . . OFF
10. Fire pull handle . . . . PULL
11. Climb to safe altitude; land as soon as practicable.

WARNING

Landing should not be attempted until sufficient fuel has been dumped or burned down.

BLOWN TIRE DURING TAKEOFF

Takeoff Aborted

1. Maintain directional control.
   a. Nose Wheel Steering. . . . . ENGAGED
   b. Anti-skid switch . . . . OFF
   c. Brakes . . . . AS REQUIRED
2. Follow aborted take-off procedures.

Takeoff Continued

1. Do not retract gear
2. Maintain airspeed below 220 KIAS
3. Hydraulic pressures. . . . . CHECK

Note

If chase is available, check for debris damage.

4. Burn or dump excess fuel and follow landing with blown tire procedures.

AFTERBURNER FAILURE DURING TAKEOFF

If an afterburner fails during takeoff, the loss of thrust is significant. Takeoff need not be aborted if take-off speed and distance are compatible with runway remaining. Afterburner relight may be attempted by retarding throttle of affected engine to MIL and readvancing to afterburner range.

FIRE WARNING LIGHT DURING TAKEOFF

1. ABORT (if feasible)
2. Throttles. . . . . IDLE

3. Affected engine . . . . OFF

4. Affected fire pull handle . . . . PULL

5. Spoiler brakes . . . . BRAKE

6. Arresting hook . . . . DOWN

If Takeoff Is Continued:

1. Good engine . . . . MAX THRUST

2. Affected engine . . . . MIL THRUST

If Fire Is Confirmed or Light Persists:

3. Affected engine . . . . OFF

4. Fire handle . . . . PULL

If Fire Is Not Confirmed:

5. Affected engine . . . . MIL THRUST

6. External stores . . . . JETTISON
Jettison stores after airborne

7. Establish safe climb speed.

Note

On a non-afterburner or single-engine takeoff, establish a safe climb speed before retracting the landing gear. Retracting the gear opens the main gear door (speed brake) to full position to allow the wheels to enter the well.

8. Fuel . . . . DUMP

9. Landing gear . . . . UP

10. Slats and flaps . . . . AS REQUIRED

11. Throttle (affected engine) . . . . OFF

12. Fire pull handle . . . . PULL

If Indications Of Fire Persist:

13. EJECT

WARNING

If the fire warning lamp illumination is accompanied by positive FIRE indications, explosion or vibration, abnormal engine instrument readings, smoke or fumes in the cockpit, burning odor in oxygen system mask, trailing smoke, or verification from another aircraft or control tower. . EJECT

If Indications Of Fire Do Not Persist:

14. Land as soon as practicable.

Note

If fire is not evident and warning lamp goes out, operate at reduced power settings and make a precautionary landing.

IN-FLIGHT EMERGENCIES

SINGLE-ENGINE FLIGHT CHARACTERISTICS

WARNING

At certain gross weights and ambient temperatures, the aircraft will NOT fly on one engine in the take-off or landing configuration.

Because of the location of the engines relative to the centerline of the aircraft, only a slight rudder deflection is required to prevent a yaw toward the failed engine. Minimum single-engine control speed varies with gross weight, flap setting, and the landing gear position. The aircraft is designed so that no one system (hydraulic, pneumatic, electrical, etc.) is dependent on a specific engine. Therefore, loss of an engine will not result in a loss of any complete system.

AIRSTART

In general, airstart capability is increased by higher airspeeds and lower altitudes. No damage will occur to the engine if starts are attempted outside the envelope. However, unsatisfactory starts on the low-speed side of the envelope may result in an increase in TIT towards the start limit (705°C) due to a hung start. Care should be taken not to exceed this limit. See AIR START ENVELOPE (figure 5-1).

1. Fuel panel/quantity . . . . CHECK

2. Throttle . . . . OFF

3. Airstart ignition button . . . . DEPRESS

4. Throttle . . . . IDLE

If Relight Is Not Accomplished Within 50 Seconds:

5. Throttle . . . . OFF

6. Airstart ignition button . . . . DEPRESS

7. Throttle . . . . IDLE

If Relight Is Not Accomplished Above 16%:

8. Throttle . . . . OFF

9. Engine start switch . . . . ENGINE FOR START

10. Throttle (affected engine) . . . . START

11. Throttle (affected engine) . . . . IDLE (at 16%)

SINGLE-ENGINE FAILURE

1. Throttle (affected engine) . . . . OFF

Note

If engine failure is something other than mechanical failure, an airstart may be attempted.

2. Land as soon as practicable.

DOUBLE-ENGINE FAILURE

Should a double-engine failure occur and an airstart of one or both engines cannot be effected, control for flight is dependent on maintaining sufficient windmill rpm on the engines.

If repeated relight attempts are not successful, eject by 2,000 feet AGL.

If still on first or second relight attempt when passing through 10,000 feet AGL, and it appears that a relight is likely, airstart attempt may be continued to a minimum of 2,000 feet AGL.

If both engines are still flamed out below 10,000 feet AGL, zoom to convert excess airspeed to altitude. Attempt a normal airstart as time permits. If the peak altitude is 2,000 feet AGL and the airstart attempt is not successful, eject no lower than 2,000 feet AGL. If the peak altitude is below 5,000 feet AGL and an airstart attempt is made during the zoom and there is no evidence of a relight, eject at the peak altitude. If no airstart attempt is made, eject at the peak altitude.

If the decision to eject is made at high altitude, it is recommended that the aircrew eject at a minimum of 2,000 feet AGL.

FIRE

ENGINE FIRE

1. Throttle (affected engine) . . . . IDLE

If Fire Is Confirmed:

2. Throttle . . . . OFF

3. Fire pull handle . . . . PULL

If Fire Continues:

4. Eject

If Fire Ceases:

5. Land as soon as practicable.

ELECTRICAL FIRE OR SMOKE IN COCKPIT

Circuit breakers protect most circuits. However, only a limited number are provided in the cockpit to isolate electrical failures. Because of the electrical complexity of the aircraft, electrical fires may occur and immediate action is required. Few electrical fires may be isolated visually and immediate corrective action is required.

Should symptoms of an electrical fire occur, and the cause cannot be determined, proceed as follows:

1. Maneuver aircraft into safe flight envelope for damper off flight.

2. Unnecessary electrical equipment and circuit breakers . . . . OFF

3. Land as soon as practicable.

SMOKE AND FUME ELIMINATION

1. Oxygen mask and flow . . . . CHECK

2. Air source selector . . . . L ENG then R ENG

If Source of Smoke Cannot Be Isolated To An Engine:

3. Air source selector . . . . OFF

4. Decelerate . . . . BELOW MACH 1.0

5. Descend to below 28,000 feet.

6. Air source selector . . . . RAM (LIMIT 320 KIAS)

If Source of Smoke Or Fumes Is Not Isolated:

7. Non-essential electrical equipment . . . . . OFF

8. Essential equipment . . . . ON
As required
Turn on electrical equipment, one system at a time, and check for smoke until source is determined.

WARNING

To prevent excessive temperatures when pressure suits are worn, the air conditioning system mode selector switch must not be placed to the OFF position prior to or while operating in RAM.

Note

- Moving the air source selector knob from OFF to RAM should be accomplished without pausing in the intermediate positions, to prevent the possible introduction of more smoke from one or both of the engines.

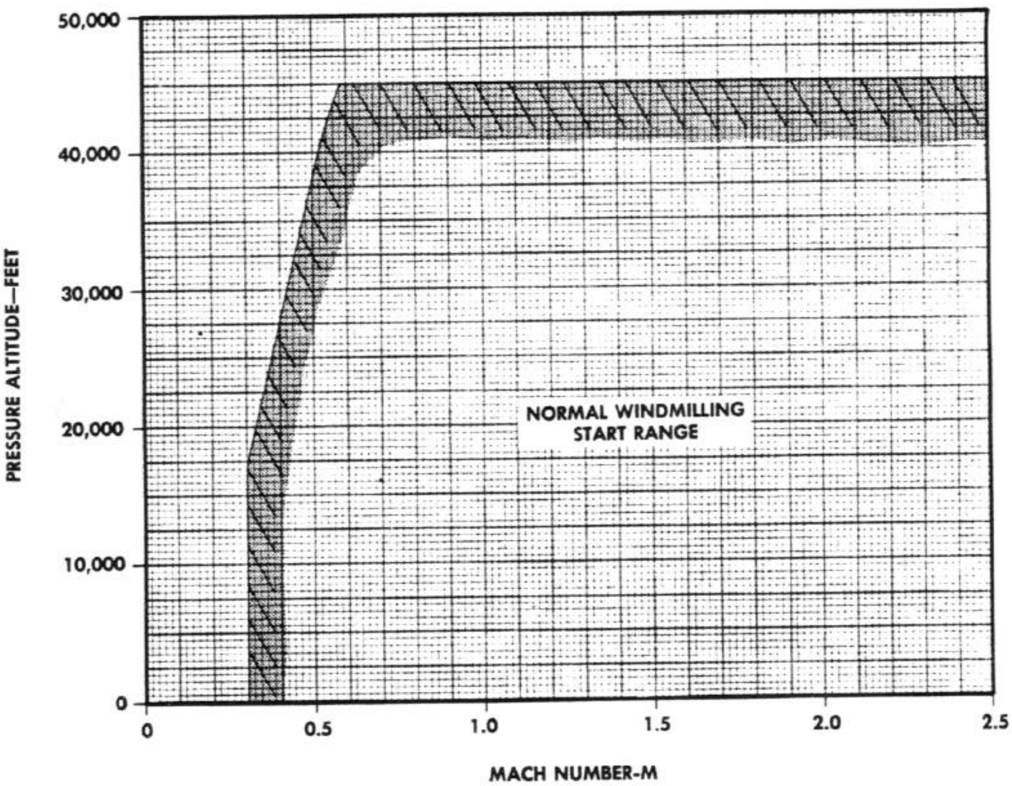
- Selecting RAM position will open the ram air scoop, dump cabin pressure, and close the pressure regulating and shutoff valve.

AIR START ENVELOPE

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINES(S): (2) TF-30-P-12
NO AIRBLEED
NO POWER EXTRACTION
RAM RECOVERY PER MIL-E-50088
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/126-0

Figure 5-1

UNLOCKED CANOPY INDICATION

1. Visually check canopy handle position.

2. Decelerate to below 250 KIAS and descend below 8,000 feet if practical.

Note

Past experience indicates that in-blast and noise are not severe at 250 KIAS and below. In flight, the canopy will remain between full and half open and slam shut on touchdown.

3. Pressurization switch . . . . . . AS REQUIRED

   a. Combat (above 30,000 feet)

   b. Dump (below 30,000 feet)

4. Visors . . . . DOWN

5. Seat. . . . . DOWN

6. Oxygen mask and fittings . . . . CHECK
   Check for security.

WARNING

When the cabin pressure schedule is changed from normal to combat, monitor the cabin pressure altimeter for a rapid increase in cabin altitude. If the cabin altitude does not increase, immediately position the pressurization selector switch to DUMP.

7. Canopy latch handle . . . . . . CHECK LOCKED

8. Land as soon as practicable.

GLASS PANEL CRACKS OR FAILURE

1. Decelerate and descend.

2. Visors . . . . DOWN

3. Seats . . . . DOWN

4. Oxygen mask and fittings . . . . CHECK
   Check oxygen mask and oxygen hose fittings for security.

5. Land as soon as practicable.

COMPRESSOR STALL

A compressor stall is an aerodynamic disruption of the airflow through the compressor and is caused by subjecting the compressor to a pressure ratio above its capabilities at the existing conditions. Compressor stalls may be induced by engine or inlet control malfunction, excessive angle-of-attack or yaw causing poor inlet air distribution, or rapid throttle reversal (high power to low power and return).

Compressor stalls may be self-clearing, may cause flameout, or may result in a steady state, fully developed stall. In the first case no immediate action is required. In some cases the engine will stall and immediately recover with only an evidence of a stall being a light-to-moderate "bang". In the second case the automatic restart circuit in the engine will furnish ignition and the engine may be recovered by moving the throttle to idle to gain a restart and then reapplying power. The third case requires recognition and corrective action to restore power and prevent damage to the engine from overtemperature. A compressor stall may be recognized by a pulsation felt through the airframe, an audible noise which may vary from a faint muffled thud to a very loud "bang", a loss of thrust indicated on the engine instrument, no EPR response to throttle movement and, as a general rule, a rise in turbine inlet temperature. In the event of compressor stall on one or both engines, proceed as follows:

1. Throttle (affected engine(s)) . . . . IDLE
   Move the throttle of the affected engine to IDLE and check for recovery. If the engine recovers, attempt gradual application of power. If supersonic, advance throttle to MIL or above.

If Stall Will Not Clear:

2. Throttle (affected engine). . . . . OFF

3. Initiate airstart.

4. When engine recovers, set power as desired.

Note

If a compressor stall or afterburner blowout occurs in afterburner operation, but a fully stalled engine condition does not follow, an afterburner relight from military power may be attempted immediately at any flight condition.

EJECTION

Every emergency in which ejection is considered will have its particular set of circumstances, involving such factors as speed, attitude and control, and altitude. Under level flight conditions, eject at least 2,000 feet above the terrain whenever possible.

WARNING

Do not delay ejection below 2,000 feet above the terrain if repeated relight attempts are not successful or for other reasons that may commit you to marginal conditions for safe ejection. Accident statistics emphatically show a progressive decrease in successful ejections as altitude decreases below 2,000 feet above the terrain.

Note

- Ejection above 700 KIAS is not recommended.
- Above 50,000 feet manually select emergency pressurization before ejecting.

Under spin or dive conditions, eject at least 15,000 feet above the terrain whenever possible. If the aircraft is controllable, attempt to decelerate as much as practical prior to ejection by zooming the aircraft, thus trading airspeed for altitude. If the aircraft is not controllable, ejection must be accomplished at whatever speed exists. An ejection at low altitudes is facilitated by pulling the nose of the aircraft above the horizon ("zoom-up maneuver"). This maneuver affects the trajectory of the crew module, providing a greater increase in altitude than if ejection is performed in a level flight attitude. Provided a positive rate of climb is maintained, this gain in altitude will increase the time available for complete actuation of the ejection equipment. During extremely low-altitude ejections, the automatic features of the equipment must be used and depended upon. Safe ejection is enhanced by establishing the best conditions possible prior to ejection. The ejection envelope is shown in figure 5-1A. The envelope reflects only the best or safest conditions; the decision to eject or not eject in an emergency should not be rigidly determined by the fact that the aircraft is in or out of the "Known Safe" envelope.

PRE-EJECTION (IF TIME PERMITS)

1. Throttles . . . . RETARD
   Trade airspeed for altitude.
2. Advise MCO of situation.
3. Transmit MAYDAY (give position).
4. IFF . . . . EMERGENCY
5. Inertia reel . . . . LOCKED
6. Chaff interrupt lever . . . . . Select ON or OFF
   If not previously selected.

EJECTION

Note

Locking trigger must be held fully depressed or handle may not PULL.

1. Ejection handle . . . . SQUEEZE and PULL

DESCENT

1. Emergency oxygen handle . . . . PULL (if required)
2. Emergency pressurization ring . . . . . . PULL (if required)

When Below 15,000 Feet:

3. Parachute deploy handle . . . . PULL (if required)

WARNING

After ejection, keep O_2 mask on or visor down on pressure suit-type helmet to avoid breathing noxious or toxic fumes that will be present in the module. After main chute deployment, canopy hatches may be opened for ventilation but should be closed before landing. Air ventilation masks should be used after emergency oxygen supply is depleted.

AFTER GROUND LANDING

1. Severance and flotation handle . . . . . . PULL
2. Parachute release handle . . . . PULL
3. Restraint harness buckles, upper and lower . . . . RELEASE
4. Oxygen mask hose . . . . DISCONNECT
   With pressure suit, disconnect oxygen inlet hose from suit controller.
5. Canopy hatch . . . . OPEN

Note

Subsequent step depends upon decision to evacuate crew module or remain in it as a survival shelter.

6. Survival equipment . . . . REMOVE
   Access instructions are located behind right seat back cushion.

AFTER WATER LANDING

1. Severance and flotation handle . . . . PULL
2. Parachute release handle . . . . PULL
3. Oxygen mask . . . . REMOVE
4. Restraint harness buckles, upper and lower . . . . RELEASE
5. Bilge pump . . . . ENGAGE and OPERATE (if required)
6. Auxiliary flotation handle. . . . . PULL
7. Canopy hatch . . . . OPEN

Note

If inclement weather prevents opening canopy hatch, utilize air ventilation mask in compartment behind right seat.

8. Survival equipment . . . . REMOVE
   Access instructions are located behind right seat back cushion.

CREW MODULE EJECTION ENVELOPE

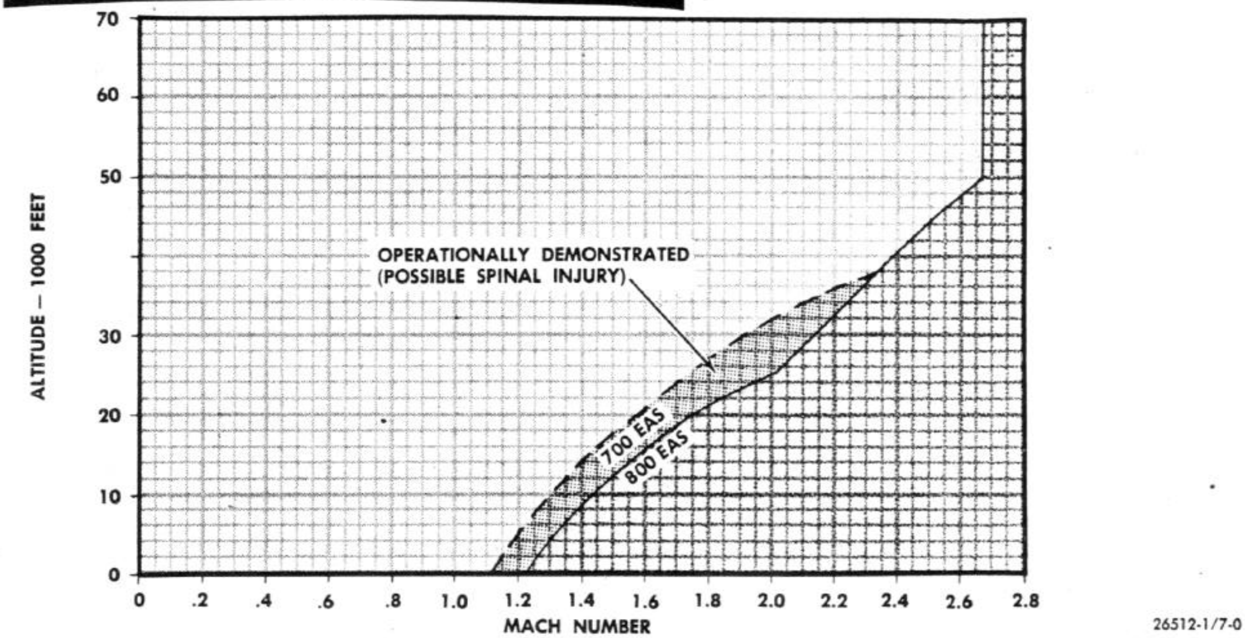


Figure 5-1A

HELICOPTER RESCUE FROM CREW MODULE IN WATER

1. Canopy latch handle. . . . . .PULL OUT
   Pull out to first detent position to lock counterpoise.

WARNING

If counterpoise is not locked, the piston will extend rapidly when released in following step and can cause injury.

2. Counterpoise attach pin. . . . . .REMOVE
   Remove pin while pushing up on canopy hatch to relieve weight on pin.

3. Canopy seal pressure hose. . . . DISCONNECT

4. Canopy detach initiator safety pin . . .REMOVE

WARNING

Helmet must be worn to prevent hearing damage from the explosion of the ballistic devices generated during canopy removal.

5. Flight helmet . . . . DON

Note

In heavy seas, pull auxiliary flotation handle to allow additional freeboard.

6. Canopy detach handle. . . . . .PULL

7. Canopy hatch . . . . PUSH OVERBOARD

ELECTRICAL SYSTEM FAILURE

SINGLE GENERATOR FAILURE

Failure of one generator will be noted by the illumination of the applicable caution lamp. One generator in normal operation is sufficient to support the entire electrical load or demand. Should generator caution lamp illuminate, proceed as follows:

1. Electrical control panel . . . . CHECK
   Check electrical control panel for TIE indication in power flow indicator.

2. Generator switch. . . . . OFF, then ON

3. Generator caution lamp . . . . CHECK
   If the generator fault has been corrected, the generator will be reconnected to the system and the caution lamp will go out. The power flow indicator will indicate NORM.

If Generator Caution Lamp Remains Illuminated:

4. Generator switch. . . . . .OFF then TEST
If the caution lamp goes out in TEST position, the generator is operating normal and the malfunction is associated with the contactor circuit or the caution lamp circuit. The generator switch should be returned to OFF.

If Caution Lamp Remains Illuminated In TEST:

5. Generator switch. . . . . .OFF

6. Generator decouple button . . . . DEPRESS

DOUBLE GENERATOR FAILURE WITH BOTH ENGINES OPERATING

Double generator failure will not result in a total loss of electrical power for more than the 3 seconds required for the emergency generator to provide power for the essential BUS.

Note

In the event that the emergency generator does not come on within three seconds, place the emergency generator switch to ON.

1. Electrical control panel . . . . CHECK
Check electrical control panel for EMERG indication in power flow indicator.

2. Generator switches . . . . OFF, then ON

3. Unnecessary electrical equipment. . . . . .OFF

4. Maintain 1 g flight.

WARNING

To assure adequate hydraulic pressure for emergency generator operation, do not open speedbrake. Maintain a minimum of 90 percent rpm on both engines while closing speedbrake, if it is open, and for wing sweep and landing gear extension.

5. Fuel panel . . . . CHECK

6. Either engine fuel feed or fuel transfer selector .OFF

7. Land as soon as practicable.

EMERGENCY GENERATOR OPERATION WITH ONE ENGINE SHUTDOWN

Note

If the emergency generator does not come on within 3 seconds, place the emergency generator switch to ON.

1. Establish and maintain 1 g flight and an airspeed of 350 KIAS. Then, maintain a minimum of 90 percent rpm on the operating engine.

2. Do not open or close speedbrake.

3. Sweep wings forward to 26 degrees by moving the wing sweep handle at a smooth rate not to exceed 1 degree of sweep per second.

Note

Flight control damper transients may be experienced if hydraulic demands cause an interruption of the emergency generator power.

4. Landing gear. . . . . .EXTEND
Extend the landing gear, using EMERGENCY LANDING GEAR procedures.

CAUTION

Leave the gear handle in the UP position until the landing gear is down.

5. Land as soon as practicable.

Note

When operating on emergency electrical power, engines may be operating on suction feed. Use of afterburners should be avoided above 6,000 feet and military thrust above 30,000 feet.

FUEL SYSTEM EMERGENCY OPERATION

FUEL SYSTEM OPERATION ON EMERGENCY ELECTRICAL POWER

Operating on the emergency generator will operate only one fuel booster pump at a time (number 4 pump in the forward tank or number 5 pump in the aft tank) or the two outboard wing transfer pumps. The transfer pumps cannot be operated while one of the fuselage booster pumps is operating. When the engine feed selector switch is in FWD, only the number 4 pump in the forward tank will be operating and will supply fuel to both engines. When the engine feed selector switch is in AFT or BOTH, only the number 5 pump in the aft tank will be operating and will supply fuel to both engines. When the engine feed selector switch is in AUTO, either pump 4 or pump 5 will operate, depending on fuel distribution.

If the fuel differential is greater than 5,850 pounds, number 4 pump will supply fuel to the engines. If the fuel differential is less than 5,350 pounds, number 5 pump will supply fuel to the engines. If, when the AUTO position is initially selected, the fuel differential is less than 5,350 pounds, the number 5 pump will transfer fuel to the forward tank until the proper fuel differential is established. From this point on, either pump 4 or 5 will be automatically selected to supply fuel directly to the engines. During the period that pump 5 is transferring fuel forward, the engines

will be operating on suction feed. During suction feed, the fuel manifold low pressure caution lamps may come on. To transfer fuel from the wing tanks, the engine feed selector switch must be OFF and the fuel transfer switch placed to WING. This will result in the engines being fed by suction from the forward tank. Fuselage tank fuel quantities must be closely monitored to maintain the proper distribution during wing transfer. If distribution gets out of tolerance, it can be corrected by turning the wing fuel transfer switch to OFF and positioning the engine feed selector switch to AUTO.

ENGINE FEED

1. Engine fuel feed selector . . . . AUTO
   Closely monitor fuel quantity in the fuselage tanks to maintain 5600 (±250) pounds fuel differential.
2. Fuel tank pressurization selector switch . . . . PRESSURIZE

FUEL TRANSFER

1. Engine fuel feed selector . . . . OFF
   Monitor fuel quantity in the fuselage tanks to maintain 5600 (±250) pounds fuel differential.
2. Fuel transfer selector . . . . .WING

Note

When the wings are swept aft, a larger amount of fuel will be trapped in the wing tanks. To transfer all available fuel from the wing tanks, the wings must be in the extended positions. Gravity transfer of fuel is not possible.

FUEL MANIFOLD LOW-PRESSURE CAUTION LAMP INDICATION

One Fuel Manifold Low Pressure Caution Lamp

1. Engine fuel feed selector . . . . .CHECK
2. Flowmeter. . . . . .CHECK
   Check appropriate fuel flowmeter to determine if fuel flow is excessive.

If Fuel Flow Is Excessive:

3. Affected engine . . . . .SHUT DOWN
4. Fire handle . . . . PULL
5. Speed brake . . . . .EXTEND

If Fuel Flow Is Normal:

6. Totalizer . . . . .CHECK
   Check totalizer for excessive loss of fuel.

If Fuel Loss Is Excessive:

7. Perform steps 3, 4, and 5.

If Fuel Consumption Is Normal:

8. Fuel pump low-pressure advisory lamps. . . . . CHECK

If Fuel Pump Low-Pressure Advisory Lamps Are On: (Indicates fuel pump failure.)

9. Throttle (affected engine) . . . . AS REQUIRED
   Observe operating limits for fuel pump-out operation.

Note

With all fuel pumps inoperative and normal tank pressurization available, the engines will operate in maximum afterburner up to 6,000 feet, then decrease to military power up to 30,000 feet.

If Fuel Pump Low-Pressure Advisory Lamps Are Normal:

10. Descend below 30,000 feet and land as soon as practicable.

Two Fuel Manifold Low-Pressure Caution Lamps

1. Engine fuel feed selector knob. . . . . . CHECK
   Check engine fuel feed selection to insure that fuel is available to the engines.
2. Flowmeters . . . . CHECK
   Check fuel flowmeters to determine if fuel flow to either engine is excessive.

If Either Flowmeter Indicates Excessive Flow:

3. Engine affected. . . . . SHUT DOWN
4. Fire handle. . . . . PULL
5. Speed brake . . . . EXTEND

If Fuel Flow Is Normal:

6. Totalizer. . . . . CHECK
   Check to determine if there is an excessive loss of fuel.

If Fuel Loss Is Excessive:

7. Throttles. . . . . RETARD
   Retard throttles until one caution lamp goes out.
8. Perform steps 3, 4 and 5 for engine with caution lamp still on.

If Fuel Consumption Is Normal:

9. Fuel pump low-pressure advisory lamps . . . . CHECK

If Fuel Pump Low-Pressure Advisory Lamps Are On: (Indicates fuel pump failure.)

10. Throttles. . . . . AS REQUIRED
    Observe operating limits for boost-pump-out operation.

Note

With all fuel pumps inoperative and normal tank pressurization available, the engines will operate in maximum afterburner up to 6,000 feet, then decrease to military power up to 30,000 feet.

If Fuel Booster Pump Low-Pressure Advisory Lamps Are Normal:

11. Descend below 30,000 feet and land as soon as practicable.

OIL SYSTEM MALFUNCTION

An oil system malfunction is recognized by abnormal oil pressures, low oil quantity, a complete loss of oil pressure, or excessive oil temperature. If an oil system malfunction has caused prolonged oil starvation of engine bearings, the result will be a progressive bearing failure and subsequent engine seizure. This progression of bearing failure starts slowly and will normally continue at a slow rate up to a certain point at which the progression accelerates rapidly to complete bearing failure. The time interval from the moment of oil starvation to complete failure depends on such factors as: Condition of bearings prior to the starvation; operating temperatures of bearings; and bearing loads. Bearing failure due to oil starvation is generally characterized by a rapidly increasing vibration. When the vibration becomes moderate to heavy, complete seizure will occur in seconds. In order to minimize engine damage and conserve remaining operating time for possible emergencies, the affected engine should be shut down upon first recognition of an oil system failure. Upon first recognition of sustained oil system failure (above or below oil pressure limits), perform the following:

OIL PRESSURE BETWEEN 30 AND 40 PSI (EXCEPT AT IDLE)

1. Throttle (affected engine) . . . . IDLE

2. Monitor oil pressure.

OIL PRESSURE BELOW 30 PSI

1. Throttle (affected engine) . . . . OFF
   If flight conditions permit.

CAUTION

If oil pressure goes to below 30 psi and it is necessary to keep the engine operating to sustain flight, engine seizure can be expected.

OIL PRESSURE ABOVE 50 PSI

1. Throttle (affected engine) . . . . .RETARD
   If oil pressure can be maintained in 40 to 50 psi range, continue to operate engine at reduced power. If oil pressure cannot be reduced to the 40 to 50 psi range, shut the engine down.

EXCESSIVE OIL TEMPERATURE FOLLOWING POWER REDUCTION

Should an engine oil hot caution lamp illuminate following a rapid thrust reduction, it is recommended that the throttle be advanced to a higher setting if possible. This will increase the fuel flow to the cooler and increase the cooling capacity of the oil cooling unit until the heat rejection from the engine can be accommodated by lower fuel flow through the fuel oil cooler. Retarding the throttle will not normally reduce the oil temperature. If the oil hot caution lamp does not go out within 1 minute, the engine should be shut down.

SPIKE SYSTEM FAILURE

Since there is no positive means of determining spike position, a spike system failure or spike mispositioning can be recognized only by a reduction in engine or engine inlet performance. The evidence of a spike system failure will differ according to airspeed at the time of failure. Failure of the spike system at mach numbers above 1.5 will most probably be evidenced by inlet buzz or compressor stall or both. Failure of the spike in the lower speed range may result in an engine compressor stall.

1. Airspeed. . . . . REDUCED
   If above mach 1.5 and inlet buzz or compressor stall is present, decelerate until buzz or compressor stall disappears.

LANDING

1. Engine spike caution lamps . . . . OUT
   When at mach 0.3 (approximately 200 KIAS at sea level or 175 KIAS at 6,000 ft), check that engine spike caution lamps are out.

If Either Lamp Is Illuminated:

2. Applicable spike control switch. . . . . . ORIDE

CAUTION

If spike switch is placed in ORIDE, do not move out of ORIDE or loss of utility hydraulic pressure will result.

If Lamp Remains Illuminated:

3. Fly a traffic pattern that will preclude the need for high power settings for landing.

ENVIRONMENTAL CONTROL SYSTEM MALFUNCTION

If uncontrolled cabin overheat occurs, place the air source selector knob to OFF and turn off all non-essential electric equipment until below 28,000 feet and 320 KIAS or mach 1.0, whichever is lower. Thereafter the RAM position may be selected. Disconnect suit ventilation hose if in use. Land as soon as practicable.

FLIGHT CONTROL SYSTEM MALFUNCTIONS

Various flight control system malfunctions are indicated by the illumination of an associated caution lamp. All system malfunctions, however, do not constitute a potential emergency, even though the associated caution lamp is illuminated. Therefore, only those malfunctions which may develop into an emergency are covered here.

PITCH OR ROLL GAIN CHANGER CAUTION LAMP ILLUMINATED

An error in one of the redundant gain changers will cause the pitch or roll gain changer caution lamp to illuminate.

1. AFCS reset button. . . . . DEPRESS

If Lamp Does Go Out:

2. Continue normal operation.

If Lamp Does Not Go Out:

3. Airspeed. . . . . BELOW 320 KIAS or MACH 0.8

PITCH, ROLL, OR YAW CHANNEL CAUTION LAMP ILLUMINATED

Failure of one of the redundant electrical signal paths causes the appropriate channel caution lamp to illuminate. The failed signal will be electronically rejected and damping will be unaffected. If failure was a zero command, the appropriate channel lamp will come on during a maneuver. Depressing the AFCS reset button will cause the caution lamp to go out for this type of failure. Normal operation can be continued as long as the channel lamp can be reset since any subsequent failure will cause either no affect or zero stability augmentation in the affected channel. If the failure is a hard-over signal, the system electrically rejects the failed signal and the channel lamp will immediately illuminate and will not go out. For this condition, normal damping is present; however, a secondary failure could cause a hard-over damper servo. Airspeed should be reduced to the stability augmentation off limits and the affected damper turned off. The aircraft should be landed as soon as practicable.

1. AFCS reset button. . . . . DEPRESS

If Lamp Does Go Out:

2. Continue normal operation.

If Lamp Does Not Go Out:

3. Affected damper switch. . . . . OFF
   Within stability augmentation off limits.

4. Land as soon as practicable.

PITCH, ROLL, OR YAW DAMPER CAUTION LAMP ILLUMINATED

Illumination of a damper lamp indicates the three signals to the damper servo do not agree. If the lamp remains out after the damper reset button is momentarily depressed, one of the three signals has failed to a zero or null command. Any subsequent failure will result in either normal operation or zero damping. If the damper lamp remains illuminated after the AFCS reset button is momentarily depressed, one of the three signals has failed to a hard-over command and has been voted out. A subsequent failure in that axis could cause the damper to go hardover. Certain power failures to the flight control computers have the affect of causing one damper command to fail to a zero or null command. These cases should be treated the same as a damper lamp that will reset. If a damper lamp illuminates, proceed as follows:

1. Airspeed . . . . REDUCED
   Reduce speed to the applicable stability augmentation off limits.

2. AFCS reset button . . . . .DEPRESS MOMENTARILY

If Lamp Does Go Out:

3. Continue normal operation.

If Lamp Does Not Go Out:

4. Affected damper . . . . OFF

5. Land as soon as practicable.

RUDDER AUTHORITY CAUTION LAMP ILLUMINATED

If rudder authority differs from that programmed by the control system switch the rudder authority caution lamp will illuminate.

1. Rudder authority switch . . . . .CHECK
   Check that the rudder authority switch is in AUTO. If lamp remains illuminated, the rudder authority may be unscheduled.

CAUTION

At high speeds, exercise caution in the use of rudder pedals. For landing, if lamp remains illuminated, place the rudder authority switch to FULL. If the lamp still remains on, rudder and nose wheel steering authority may be limited.

HYDRAULIC SYSTEM FAILURE

Failure of either hydraulic system will cause the pitch, roll, and yaw damper caution lamps and the hydraulic low-pressure caution lamps to illuminate. The damper servo actuators will operate as non-redundant servos. As the hydraulic pressure drops and the damper caution lamps illuminate, forces may be felt in the control stick.

Supersonic

1. Throttles. . . . . .RETARD
   Reduce airspeed to damper off envelope.

2. Control system switch. . . . . STBY

3. Wing sweep handle. . . . . 50 DEGREES
   Maintain 1.0 g during sweep.

CAUTION

Move the wing sweep handle at a smooth rate not to exceed 1 degree of sweep per second to avoid depleting hydraulic pressure.

Note

Do not attempt to reset dampers.

4. Pitch damper . . . . OFF, IF DESIRED

Subsonic

1. Control system switch. . . . . STBY

2. Wing sweep handle. . . . . .EXTEND
   Maintain wing sweep position compatible with airspeed and sweep wings to 26 degrees as soon as practicable. Minimize flight control movement during wing sweep and speed brake operation.

3. Land as soon as practicable.

COMPLETE HYDRAULIC SYSTEM FAILURE

If complete hydraulic system failure occurs, all normal flight controls will be inoperative. Upon initial detection of hydraulic pressure loss or gage fluctuation, reduce airspeed and attempt to establish level flight. If aircraft is uncontrollable, EJECT.

Note

Partial control of the aircraft may be maintained with a complete hydraulic system failure by use of symmetric and differential thrust.

EMERGENCY WING SWEEP OPERATION

The necessity for emergency wing sweep operation may arise from either of two conditions: one engine inoperative, or one hydraulic system inoperative. In either condition, normal wing sweep commands can result in a severe drop in available hydraulic pressure, thus degrading flight control response. While sweeping the wings under the above conditions, maintain as high an rpm on the engine(s) as practicable, and maintain 1 g straight and level flight. When operating with one engine, if the operating engine rpm is allowed to drop below 90 percent and the windmilling engine is below 40 percent rpm, it will be necessary to sweep the wings in increments of 1/4 to 1/2 inch of wing sweep handle movement.

Supersonic

1. Wing sweep handle. . . . . 50 DEGREES
   If an engine failure or hydraulic system failure occurs at supersonic speed, sweep the wings as soon as practicable.

Subsonic

1. Wing sweep handle. . . . . 26 DEGREES
   When subsonic, sweep the wings while maintaining straight and level 1 g flight.

LANDING EMERGENCIES

WARNING

Unlock the canopy latch handle lock tab prior to all landings to ensure access to cockpit, if needed, for crew rescue on the ground.

PRIMARY OR UTILITY HYDRAULIC SYSTEM FAILURE

Fly an extended downwind leg to provide time for lowering the landing gear and flaps by the alternate method. After touchdown, normal braking will be available until the brake accumulator pressure has been reduced to 1100 ± 100 psi (after approximately 18 full-brake applications). Differential braking must be used to maintain directional control during landing roll. To minimize consumption of brake accumulator hydraulic fluid, braking should be accomplished by as few brake applications as possible. A single moderate and steadily increasing brake application is recommended. If the number of brake applications used is great enough to reduce accumulator pressure to less than 1100 ± 100 psi, normal braking will not be available and it will be necessary to pull the auxiliary brake handle to stop the aircraft.

1. Anti-skid. . . . . OFF

2. Do not attempt to reset damper caution lamps, if illuminated.

3. Landing gear. . . . . .EXTEND
   Extend the landing gear, using landing gear emergency extension procedures.

4. Spike control switches. . . . . . AS REQUIRED
Prior to landing with either hydraulic system inoperative, decelerate to mach 0.3 (approximately 200 KIAS at sea level or 175 KIAS at 6,000 ft), and check that engine spike caution lamps are out. If either lamp is illuminated, place the affected spike control switch to ORIDE.

CAUTION

Do not move spike switch out of ORIDE or loss of utility hydraulic pressure will result.

5. Slats and flaps . . . . EXTD
Use emergency slats/flaps extension procedures.

6. Maintain directional control after touchdown by differential braking.

Note

- With only the utility hydraulic system operative, only the outboard spoilers will be available. If only the primary hydraulic system is operative, only the inboard spoilers will be available.
- Lateral response rates will be reduced when one pair of spoilers is inoperative.

MAIN LANDING GEAR FAILURE TO EXTEND AND LOCK AFTER RELEASING FROM UPLOCK

1. Landing gear handle. . . . . RECYCLE
Recycle the landing gear handle from DN to UP, then back to DN. Check for gear down indication. Check circuit breakers.

2. Impose a g load on the aircraft and check for gear down indication.

3. Follow emergency landing gear extension procedure.

4. Speed brake switch . . . . OFF

5. Utility hydraulic isolate switch . . . . HOLD in ISOL

6. Landing gear handle . . . . UP

7. Speed brake switch . . . . OUT
Hold switch in OUT until 100% indication.

8. Landing gear handle . . . . DN

9. Impose a g load on the aircraft and check for a gear down indication.

LANDING GEAR EMERGENCY EXTENSION

If the landing gear cannot be lowered, using the normal procedures, or if some other system failure requires emergency gear extension to be used, proceed as follows:

1. Landing gear handle. . . . . UP

Note

Placing the landing gear handle UP is necessary for compatibility with other emergencies such as landing with one hydraulic system out, emergency generator operation with one engine shutdown, etc. If hydraulic and electrical systems are normal, emergency gear extension may be made with the landing gear handle positioned to DN.

2. Alternate gear down handle. . . . . PULL
After pulling the emergency release handle, allow time, as practicable, for the gear to fully extend.

3. Landing gear handle. . . . . DN

4. Landing gear position indicator . . . . . DOWN

5. Landing gear handle warning lamp . . . . . OUT

CAUTION

Before removing electrical power from the aircraft, push the alternate gear down handle IN. Pressure in the speed brake actuator will drive the speed brake to full open position, causing damage from ground contact.

Note

After the landing gear emergency release handle is pulled, nose wheel steering will be inoperative and the nose wheel will be cocked to one side. This will present no directional control difficulty on touchdown since the hydraulic pressure holding it cocked is slight. During landing roll, the nose wheel should be held off the runway as long as possible.

If Speed Brake Fails To Retract:

Indicated by gear handle lamp illuminated.

6. Alternate gear down handle . . . . IN

Note

- Reduced air loads during landing will allow the speed brake to extend and drag the landing surface.
- The tail bumper will not extend with alternate extension of the landing gear.

LANDING WITH UNSAFE GEAR INDICATION

1. Slow aircraft to appropriate limit speed and extend flaps to 15 degrees.

2. Landing gear circuit breakers . . . . . CHECK
   Check the landing gear controls and landing gear warning circuit breakers.

3. Landing gear handle. . . . . RECYCLE

If Landing Gear Is Still Unsafe:

4. Alternate gear down handle . . . . PULL

Note

After the landing emergency gear handle is pulled, nose wheel steering will be inoperative and the tail bumper will not extend.

If Landing Gear Is Still Unsafe:

5. Obtain a visual gear check from another aircraft or the control tower.

LANDING WITH NOSE GEAR UP OR UNLOCKED, MAIN GEAR DOWN

1. Landing gear handle. . . . . DN
2. External stores . . . . JETTISON
3. Excess fuel . . . . DUMP or BURN
   Reduce fuel to 1500 pounds.
4. Spoiler brake switch . . . . OFF
5. Anti-skid switch . . . . OFF
6. Shoulder harness . . . . LOCKED
7. Fly a normal pattern and flare the landing.
8. Throttles . . . . OFF
   Immediately after touchdown.
9. Fire pull handles . . . . PULL
10. Abandon the aircraft after coming to a full stop.

LANDING WITH MAIN GEAR OR ALL GEAR UP

Landing with the main gear or all gear UP is not recommended . . . . EJECT

LANDING WITH MAIN GEAR UNSAFE INDICATION

1. Landing gear handle. . . . . DN
2. Circuit breakers . . . . CHECK
3. Alternate gear down handle . . . PULL
4. External load . . . . JETTISON
5. Dump or burn excess fuel.

Note

If dumping operation is necessary during afterburner operation, the fuel may ignite behind the aircraft. This should cause no concern since the fire will remain behind the aircraft. Other aircraft in the immediate vicinity should be advised to stay well clear during dumping operations.

6. Anti-skid switch . . . . ON
7. Spoiler switch . . . . BRAKE
8. Shoulder harness . . . . LOCKED
9. Arresting hook . . . . DOWN
10. Fly a normal landing pattern and make a normal landing.

Note

Attempt to touchdown at normal landing attitude. Do not hold the aircraft off the runway by increasing angle-of-attack.

CAUTION

Do not taxi aircraft or shut down engines until ground lock pins are installed and the alternate gear down handle has been pushed IN.

LANDING WITH A BLOWN TIRE

Main Gear Tire

1. External stores . . . . JETTISON
2. Fly a normal landing pattern.
3. Spoiler brake switch. . . . . OFF
4. Arresting hook . . . . DOWN
5. Anti-skid switch. . . . . OFF
6. Lower nose and use nose wheel steering and brakes as required to keep aircraft on the runway.
7. If suitable arresting gear is available, fly into the gear on centerline, otherwise touchdown on side of runway opposite the blown tire.
8. Use brakes as required to maintain directional control and decelerate aircraft.

Nose Gear Tire

Use the same procedure as main gear tire except there is no need to jettison external stores.

Land in the center of the runway and hold the nose off the runway as long as possible. Anti-skid should be ON.

NOSE WHEEL STEERING MALFUNCTION

If nose wheel steering malfunction is indicated by hard-over nose wheel steering, or loss of directional control, disengage nose steering by pulling the alternate landing gear down. Maintain directional control with rudder and differential braking.

SINGLE-ENGINE LANDING AND GO-AROUND

During single-engine operation, utility and primary hydraulic system flow is reduced by almost 50 percent. Because of this, the landing gear system, speedbrake system, and wing sweep system will each absorb total flow of the utility hydraulic system. Avoid operation of more than one utility hydraulic system function at a time. Since the flight control system utilizes both utility and primary pressure, operation of necessary utility hydraulic system functions should be accomplished while in level flight. Changes in wing sweep should be accomplished in straight and level flight. Wing sweep changes may require as much as 20 seconds for completion. During wing sweep operation, no other demands should be placed on the utility system such as speed brakes, air inlet control or flaps. During the landing approach, keep the operating engine rpm as high as practicable until touchdown.

If a go-around is necessary, advance the throttle of the operating engine and continue approach until go-around airspeed is reached. When landing in a gusty crosswind, final approach airspeed should be increased by ten knots.

EMERGENCY EXTENSION OF SLATS/FLAPS

1. Airspeed. . . . . 210 KIAS or LESS
2. Slats/flaps switch . . . . EMERG
3. Emergency slats/flaps switch . . . . . EXTD
   Hold the emergency slats/flaps switch in EXTD until the slats are down and the flaps are in the desired position. Emergency extension of the flaps to full down requires 60 seconds.

NO SLATS/FLAPS LANDING

A no slats/no flaps landing is basically the same as a normal landing except the pattern is expanded and approach speeds are slightly higher than normal.

1. Flight control switch. . . . . T.O. & LDG

   Note

   If slats have not extended, flight control system will not automatically assume a take-off and landing configuration.

2. Excess fuel . . . . DUMP or BURN
   Reduce fuel to 1,500 pounds.
3. Landing check list . . . . COMPLETE
4. Approach speed. . . . . AS REQUIRED

At touchdown throttles IDLE. If runway permits, hold nose wheel off runway (approximately 10°) and begin braking when nose lowers to the runway. Hold stick full aft during rollout.

FORCED LANDING

WARNING

It is recommended that forced landings be made with the landing gear extended, regardless of the terrain. A greater injury hazard is present whenever emergency landings are made with the landing gear retracted. Increased airspeed or nose-high angle-of-attack during landings with the landing gear retracted contributes greatly to crew injuries and aircraft damage. The nose-high attitude causes the aircraft to slap the ground, subjecting the crew to possible spinal injuries. Less injury and less aircraft damage will result with the gear extended. It is not recommended that a landing on unprepared surfaces be attempted with this aircraft: the crew should. . . . . EJECT

LANDING WITH WINGS AT 26 DEGREES SWEEP OR GREATER AND NO FLAPS

Landing with wings and flaps in other than normal landing configuration will necessitate a long, shallow, straight-in approach. Avoid abrupt maneuvers or flight in excess of 1 g.

1. Excess fuel . . . . BURN or DUMP
   Because of the high approach and touchdown airspeeds involved during landing with wings swept past 26 degrees, burn or dump as much fuel as practicable prior to entering traffic pattern.

   Note

   If dumping operation is necessary during afterburner operation, the fuel may ignite behind the aircraft. This should cause no concern since the fire will remain behind the aircraft. Other aircraft in the immediate vicinity should be advised to stay well clear during dumping operations.

2. Landing check list . . . . . . .COMPLETED
3. Touchdown airspeed . . . . AS REQUIRED
   During transition from 50 feet altitude to touchdown, the rate of descent should be approximately 480 fpm. The angle-of-attack should be ________ units for wing sweep of 50° or greater and ________ units for wing sweep of 35° or less. Touchdown as near to the approach end of the runway as possible.

Note

The following braking technique is based on the assumption that sufficient runway is available. If less than the required runway is available, maximum braking should be initiated as soon as possible.

4. Throttles. . . . . IDLE

Note

Ground roll spoilers will not be available at wing sweep angles of 35° or greater.

5. After touchdown, hold the nose wheel off the runway (approximately 10° pitch attitude). At 173 KIAS, apply as much braking pressure as possible while still maintaining a 10° pitch attitude.

6. At 135 KIAS, smoothly lower the nose wheel to the runway; apply maximum braking and throttles OFF. Hold the control stick full aft to utilize the maximum drag of the horizontal tail.

CAUTION

If excessive braking is used, the wheel blowout plugs may relieve tire pressure within 3 to 15 minutes after stop. Provisions should be made to cope with wheel fire which may start shortly after the blowout plugs relieve.

7. Arresting hook. . . . . AS REQUIRED

CAUTION

Call the fire department after any emergency landing which results in hot wheels or brakes or tailhook.

8. Chock nose wheel and abandon the aircraft.

FIELD ARRESTING GEAR

The types of field arresting gear in use include the anchor chain cable, water squeezer, and Morest-type equipment. All require engagement of the arresting hook in a cable pendant rigged across the runway. Location of the pendant in relation to the runway will classify the gear as follows:

MIDFIELD GEAR. Located near the halfway point of the runway. Usually requires prior notification in order to rig for arrestment in the direction desired.

ABORT GEAR. Located 1500 to 2500 feet short of the upwind end of the duty runway and usually will be rigged for immediate use.

OVERRUN GEAR. Located shortly past the upwind end of the duty runway. Usually will be rigged for immediate use.

Some fields will have all these types of gear, others none. For this reason, it is imperative that all pilots be aware of the type, location, and compatibility of the gear in use with the aircraft, and the policy of the local air station with regard to which gear is rigged for use and when.

Speeds at which the aircraft would engage an arresting cable with arresting hook are shown on the LANDING-EMERGENCY charts in Section XI.

Note

- Under no circumstances should pilot decision to abort a takeoff be delayed because an emergency arresting gear is available at the end of the runway. Decision to abort should be based on the remaining runway and distance required for stopping, using wheel brakes. The arresting gear will then serve as an assist to stop the aircraft from rolling off the runway.
- If off center just prior to engaging arresting gear, do not attempt to go for center of runway. Continue straight ahead, parallel to centerline.

As various modifications to the basic types of arresting gear are made, exact speeds will vary accordingly. Certain aircraft service changes may also affect engaging speed and weight limitations.

Severe damage to the aircraft is usually sustained if an engagement is made in the wrong direction into the chain gear.

In general, the arresting gear is engaged on the centerline at as slow a speed as possible. Burn down to 1500 pounds or less fuel remaining. While burning down, make practice passes to accurately locate the arresting gear. Engagement should be made with the feet off the brakes, shoulder harness locked, and with the aircraft in a 3-point attitude. After engaging the gear, good common sense and existing conditions dictate whether to keep the engines running or to shut down and abandon the aircraft.
In an emergency situation, first determine the extent of the emergency by whatever means are possible (instruments, other aircraft, LSO, RDO, tower or other ground personnel). Next, determine the most advantageous arresting gear available and the type of arrestment to be made under the conditions. Whenever deliberate field arrestment is intended, notify control tower personnel as much in advance as possible and state estimated landing time in minutes. If gear is not rigged, it will probably require 10 to 20 minutes to prepare it for use. If foaming of the runway or area of arrestment is required or desired, it should be requested by the pilot at this time.

SHORT FIELD ARRESTMENTS

If at any time prior to landing, a directional control problem exists or a minimum rollout is desired, a

short field arrestment should be made and the assistance of an LSO requested. He should be stationed near the touchdown point and equipped with a radio. Inform the LSO of the desired touchdown point. A constant glide slope approach to touchdown is permitted (mirror or Fresnel Lens Landing Aid) with touchdown on centerline at or just prior to the arresting wire with the hook extended. The hook should be lowered while airborne and a positive hook-down check should be made. Use midfield gear or Morest-type, whenever available. If neither one is available, use abort gear. Use an approach speed commensurate with the emergency experienced. Landing approach power will be maintained until arrestment is assured or a wave-off is taken. Be prepared for a wave-off if the gear is missed. After engaging the gear, retard the throttle to IDLE or secure engines and abandon aircraft, depending on existing conditions.

LONG FIELD ARRESTMENTS

The long field arrestment is used when a stopping problem exists with insufficient runway remaining (i.e., aborted takeoffs, icy or wet runways, loss of brakes after touchdown, etc.). Lower the hook, allowing sufficient time for it to extend fully prior to engagement (normally 1000 feet prior to reaching the arresting gear). Do not lower the hook too early and weaken the hook point. Line up the aircraft on the runway centerline. Inform the control tower of your intentions to engage the arresting gear, so that aircraft landing behind you may be waved off. If no directional control problem exists (crosswind, brakes out, etc.), secure the engines.

ABORTED TAKEOFF

Where an aircraft takeoff must be aborted, a roll-in type engagement of all arresting gear is recommended to prevent overrun.

Note

The taxi lamp may be of use in locating arresting/abort gear at night.

BARRICADE ARRESTMENT

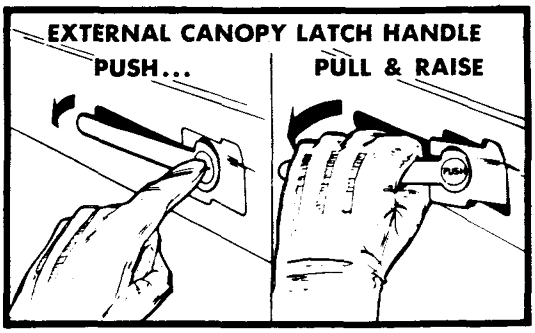
If a barricade arrestment is required, the following procedures are recommended:

1. Jettison stores if possible. Stores will not hamper successful barricade engagement, but may possibly be torn loose and present a hazard to flight-deck personnel.
2. Normally, the arresting hook should be lowered, if possible, to permit engagement of a cross-deck pendant which will minimize barricade engagement speed and damge to the aircraft, and also help to keep the aircraft on the deck at barricade entry.
3. Fly a normal pattern and approach on speed, centerline, and "meatball".
4. Anticipate loss of "meatball" for a short period of time during the approach. Barricade stanchions may obscure "meatball".

EMERGENCY ENTRANCE

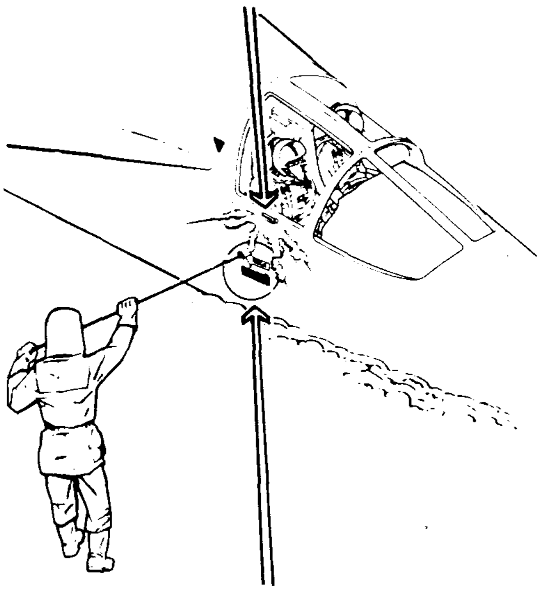
See figure 5-2.

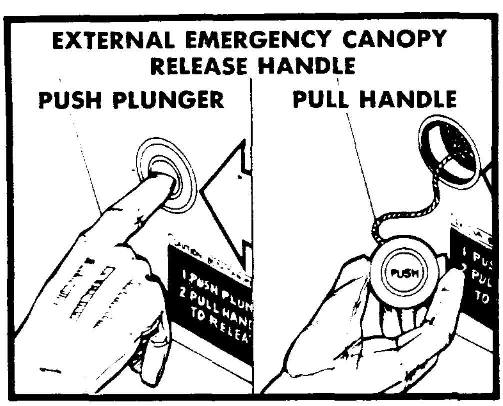
EMERGENCY ENTRANCE



NOTE

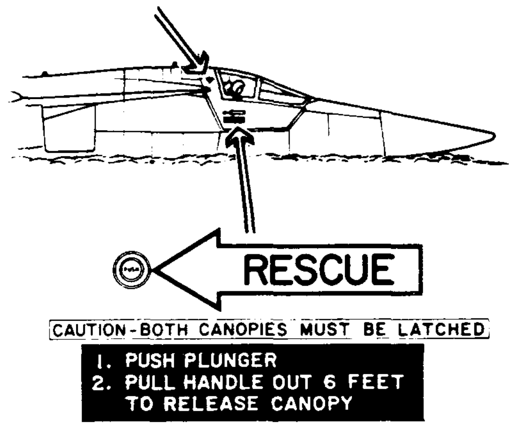
THE EXTERNAL CANOPY LATCH HANDLE MAY BE USED ONLY WHEN THE INTERNAL CANOPY LATCH HANDLE IS UNLOCKED.





NOTE

MARKING LOCATED ON LEFT SIDE OF AIRCRAFT BELOW CANOPY HATCH.



CANOPY EMERGENCY OPENING PROVISIONS ARE INSTALLED ON THE RIGHT SIDE OF THE EXTERIOR OF THE AIRCRAFT BELOW THE CANOPY HATCH.

TO RELEASE BOTH CANOPY HATCH LATCHES:

1. PUSH PLUNGER TO RELEASE SPRING-LOADED HANDLE.
2. GRASP HANDLE WHICH IS ATTACHED TO CABLE AND PULL OUT APPROXIMATELY SIX FEET.

WARNING

DO NOT PULL THE CANOPY EXTERNAL EMERGENCY RELEASE HANDLE UNLESS BOTH CANOPY HATCHES ARE CLOSED AND LATCHED. TO DO SO MAY RESULT IN FATAL INJURY TO THE OCCUPANTS FROM DEBRIS FLYING FROM THE CANOPY SILL.

CAUTION

WHEN LATCH HANDLE IS ROTATED, SILL HOOK WHICH HAS BEEN SEVERED FROM CABIN SILL MAY DROP OFF WHEN CANOPY IS RAISED.

NOTE

ACTUATING EMERGENCY RELEASE HANDLE WILL UNLOCK HATCHES ONLY. HATCHES MUST THEN BE RAISED MANUALLY.

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Figure 5-2

MASTER CAUTION LAMP ANALYSIS

Lamp Illuminated	Cause	Corrective Action
L PRI HYD R PRI HYD	Pressure output of the indicated primary hydraulic pump is below 500 ± 100 psi.	Check associated hydraulic pressure indicator. If normal indication, cross check indicator frequently for remainder of flight. If pressure loss is valid, keep hydraulic demands to a minimum and follow HYDRAULIC SYSTEM FAILURE procedures.
L UTIL HYD R UTIL HYD	Pressure output of the indicated utility hydraulic pump is below 500 ± 100 psi.	Check associated hydraulic pressure indicator. If normal indication, cross check indicator frequently for remainder of flight. If pressure loss is valid, keep hydraulic demands to a minimum and follow HYDRAULIC SYSTEM FAILURE procedures.
PRI HOT UTIL HOT	Hydraulic fluid temperature is above 115°C (240°F)	
FUEL LOW	Fuel quantity in the forward tank is less than approximately 5,000 pounds.	Check distribution and quantity of fuel supply. Transfer any other available fuel to the forward tank. Plan further flight accordingly. Select FWD position on the engine feed selector when all other fuel has been transferred to forward tank.
OIL LOW	Oil level in either the left or right engine oil supply tank drops to four (4) quarts.	Check oil quantity indicators. Cross check affected engine oil pressure. Shutdown affected engine if oil pressure starts to drop below 40 psi. Plan further flight accordingly.
INLET HOT	Inlet anti-icing air temperature (spikes and diffuser plates) excessive.	Shutoff engine anti-ice if icing lamp is not illuminated. Select AUTO again after inlet hot lamp goes out.
OXY	Oxygen quantity is 2 liters or less or oxygen pressure is less than 42 ± 2 psi.	Check oxygen quantity gage. Check cabin pressure, use emergency oxygen if necessary. Descend to safe altitude.
CABIN PRESS	Cabin altitude is above 10,000 feet.	Check position of pressurization selector switch. Check oxygen equipment and supply for normal operation.
TANK PRESS	Tank pressure is low (below 3.5 psi) and the landing gear is up or the dump switch is in DUMP. Tank pressure high (above 3.5 psi) and the landing gear is down or the refuel probe is extended.	Select tank pressure switch to PRESS Select tank pressure switch to OFF
FUEL DISTRIB	Fuel distribution is out of limits. Forward tank fuel, less aft tank fuel, is greater than 6500 pounds. Forward tank fuel, less aft tank fuel, is less than 5000 pounds.	Switch to FWD position on feed selector. Switch to AFT position on feed selector.
ICING	Icing condition sensed by ice detector.	Check probe heaters switch PRI and engine anti-ice switch AUTO. Check engine anti-icing system operation.

MASTER CAUTION LAMP ANALYSIS

Lamp Illuminated	Cause	Corrective Action
AFT EQUIP HOT	Aft electronic equipment bay temperature greater than 70° C (160° F).	When practicable, increase engine rpm to provide greater airflow. If lamp remains illuminated, turn off all nonessential electrical equipment.
WINDSHIELD HOT	Windshield temperature is above limits of 232° C (450° F).	Place rain removal switch to OFF.
L and R FUEL PRESS	Affected fuel manifold pressure is less than 15.5 psi.	Check engine fuel feed selector and fuel pump low pressure advisory lamps. Check for excessive fuel consumption. If excessive, pull FIRE PULL handle on affected engine. Observe boost pump off operating envelope.
L and R ENG OIL HOT	Affected engine oil temperature is above limits. 107° C (225° F)	If following a rapid thrust reduction, advance throttle for higher rpm. If lamp does not go out within 1 minute, shut down engine. Monitor oil pressure for above 30 psi while lamp is illuminated.
L and R ENG SPIKE	Airspeed is 0.3 mach or below and the affected spike has not extended or has not collapsed.	Affected spike control switch ORIDE.
L and R ENG OVERSPEED	Excessive N_1 rpm on affected engine.	Retard throttle of affected engine. Lamp should go out at reduced power. If lamp remains illuminated, operate at reduced power.
L and R GEN	Affected generator has malfunctioned and disconnected from the associated bus.	Check electrical control panel for power flow indication of TIE. Affected generator switch OFF, then ON. If lamp remains illuminated, follow single generator failure procedures.
ANTI-SKID	Anti-skid system has detected a malfunction and has automatically turned itself off.	Anti-skid switch OFF, then ON. If lamp remains illuminated, avoid hard braking if possible to prevent tire skids.
SPOILER	One pair of spoilers has been voted out and locked down.	Maintain positive control of aircraft attitude and decelerate to safe speed. Reset spoiler one time with neutral lateral control, but expect a rapid roll transient if spoiler is still failed. A pair of spoilers that have been voted out because of an active failure will not likely reset. The roll rate capability during landing will be reduced by approximately 50 percent.
LOW EQUIP PRESS	Pressure to forward equipment bay is less than 12.5 psi.	Turn off INS, AMCS, and radar altimeter.
PRI ATT/ HDG	Inertial navigation system is unreliable or attitude heading switch is in STBY.	If INS is unreliable, place ATT/HDG switch to STBY.
STANDBY ATTITUDE	Standby attitude system unreliable.	ATT/HDG switch to PRI.

MASTER CAUTION LAMP ANALYSIS

Lamp Illuminated	Cause	Corrective Action
TOTAL TEMP	Aircraft skin temperature is 153.3° C or more.	Reduce speed
A/B PROBE HEAT	On the ground: Probe heater switch OFF In the Air: Primary circuit has failed in one of the heaters.	On the ground: Probe Heater switch PRI In the Air: Probe heater switch SEC. If lamp remains ON, avoid icing conditions. Note The system automatically switches to secondary circuits when primary circuits fail.
BAY DOOR	Weapons bay door has not been opened automatically by the IACS.	Position emergency bay door switch to OPEN.
REF NOT ENGAGED (Lamp is directly above airspeed/ mach number indicator)	Selected autopilot reference is not engaged.	If caused by control stick steering, return stick to neutral. If caused by malfunction, depress autopilot disengage lever.
RUDDER AUTHORITY	Rudder authority differs from that programmed by the control system or, differs from that called for by slat position when control system switch is in AUTO.	Check rudder authority switch in AUTO. If the lamp remains on, the rudder authority may be unscheduled. At high speeds, exercise caution in the use of the rudder pedals. For landing, if the lamp remains illuminated, place the rudder authority switch to FULL.
CADS	One of the CADS monitors indicates malfunction.	Cross check flight instruments to determine if any are inoperative. Use standby instruments in lieu of malfunctioning primary instruments.
GAIN DIS-AGREE	Gains of AFCS are out of schedule for flight configuration, T.O. & L. configuration has not been selected by the slats switching networks.	Set CONT SYS switch to T.O. & L., if lamp illuminates with the extension of slats and flaps.
ROLL OR PITCH GAIN CHANGER	One of the triplicate roll or pitch gain changers is in error.	Depress AFCS RESET button momentarily. If the lamp goes out, continue normal operation. If the lamp does not go out, reduce airspeed and switch affected damper off. Land as soon as practicable.
ROLL, PITCH, OR YAW CHANNEL	Failure of one of the triplicate electrical signal paths.	Depress AFCS RESET button momentarily. If lamp goes out, continue normal operation. If the lamp does not go out, reduce airspeed to below 320 KIAS or mach 0.8.

MASTER CAUTION LAMP ANALYSIS

Lamp Illuminated	Cause	Corrective Action
PITCH OR YAW CHANNEL when T.O. & L. configuration is selected	One of the AYC or LSTC triplicate channels may be at fault.	Set AFCS Disconnect switch to ORIDE and depress AFCS RESET button momentarily. If the lamp goes out, proceed with landing. The AYC and LSTC functions are no longer operative. If the lamp does not go out, turn appropriate damper off.
ROLL, PITCH, OR YAW DAMPER	One of the triplicate commands to a damper servo is in error.	Depress AFCS RESET button momentarily. If lamp goes out, continue normal operation. If lamp does not go out, reduce airspeed to the stability augmentation off limits and switch affected damper off. Land as soon as practicable.

WARNING LAMPS ANALYSIS

Lamp Illumination	Cause	Corrective Action
CANOPY	Canopy hatches not closed and locked.	Close and lock canopy hatches.
CABIN PRESS	Cabin altitude is above 38,000 feet.	Descend. Check oxygen equipment.
WHEELS	Throttles are retarded to less than cruise while flaps are in a position other than fully retracted, and one of the following conditions exist: landing gear is not down and locked; speed brake is not in a trail position, or both.	Take appropriate action according to phase of flight.
REDUCE SPEED	Aircraft has flown 300 seconds in the critical temperature range of from 153.3° C to 214.3° C or that the maximum temperature index of 214.3° C has been reached or exceeded.	Reduce airspeed
RADAR ALT LOW	Aircraft falls below the preset limit index on the radar altimeter.	Climb
IFF	When in Mode 4, interrogations are not properly decoded by Mode 4 computer or replys to correctly coded interrogations are not being transmitted.	As briefed
LAUNCH BAR	On the ground: The launch bar actuator is in a position other than up. It is normal for the lamp to be illuminated during the launch sequence prior to placing either throttle to above cruise.	If lamp remains illuminated when the throttles are above cruise, do not launch - Maintenance required.
	In the air: Landing gear down and launch bar actuator is in a position other than up.	Place launch bar switch to EMERG UP

section VI
ALL WEATHER OPERATIONS

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TABLE OF CONTENTS

INSTRUMENT FLIGHT PROCEDURES

In accordance with the OPNAV Instruction 3710.7 series, the F-111B is considered a multi-piloted aircraft for instrument flight, provided both seats are occupied by currently qualified flight crewmembers.

The following procedures are general approach techniques which are modified to reflect the specific techniques used for this aircraft. Standard navigational aids include Inertial Navigation System TACAN and UHF-ADF. IFF and SIF are used for radar identification. An automatic flight control system can be used to alleviate pilot fatigue.

FLIGHT PLANNING

Clearance delays, prolonged operation at low altitudes and low airspeeds, and holding and stacking, demand a more critical approach to flight planning than would be necessary in VFR conditions. Particular attention should be paid to flight information publication and NOTAMS, and you should be thoroughly familiar with the weather and its possible changes at your destination and alternates.

Prior to entering the aircraft, a thorough pre-flight should be performed emphasizing any particular procedures needed for the existing weather conditions. If in icing conditions, pitot heat should be on.

NIGHT FLYING

The following checks and information are in addition to those given for normal instrument flight.

On entering aircraft: (External Power On)

1. Interior lighting . . . . ADJUST

Note

For night catapult launches, place instrument panel red floodlights to DIM to provide instrument panel lighting if normal instrument lights fail.

2. White floodlights . . . . CHECK
3. Navigation and exterior lights . . . . . CHECK
4. Flashlight . . . . CHECK

VFR CONDITIONS

1. Exterior lights . . . . BRIGHT (STEADY)
2. Anti-collision lights . . . . ON

During formation flights, set lights as desired.

IFR CONDITIONS

During night instrument conditions, the anti-collision lights should be turned OFF due to vertigo effect of flashing light reflections from surrounding clouds. Exterior lights should be bright and steady.

INSTRUMENT TAKEOFF

Before Taxi

Check UHF navigation for proper channel operation.

Before Instrument Takeoff

Step 2 is performed if climb-out through precipitation or clouds is anticipated.

1. Engine anti-ice switch (Recheck) . . . . . AUTO
2. Probe heaters switch . . . . ON

3. HSI heading set and course set. . AS REQUIRED
4. Take-off check list . . . . COMPLETE

Instrument Takeoff and Transition

1. Align aircraft with runway and check HSI.

INSTRUMENT CLIMB

A simplified climb schedule as outlined below may be used. However, for accurate climb schedules versus gross weight and aircraft configuration, refer to the climb charts in Section XI, Performance Data.

1. Maintain ___ to ___ nose-up attitude until reaching ___ KIAS.
2. Vary pitch attitude as necessary to maintain ___ KIAS until reaching ___ IMN.
3. Vary pitch attitude as necessary to maintain .74 IMN to cruise altitude.

For best fuel economy, military power should be used throughout the climb. However, an afterburner climb should be used for climb through moderate to severe icing.

When In The Clear, Or On Top:

4. Probe heaters switch. . . . . OFF

INSTRUMENT CRUISING FLIGHT

After leveling off from climb:

1. Establish cruise mach number.
2. Adjust power.

TURNS

Instrument turns should normally be made using 30 degrees bank.

Steep turns should be avoided if possible. If a steep turn is necessary, anticipation of the aircraft reaction becomes most important to retaining precise control.

HOLDING

Holding patterns may be flown at most altitudes at approximately 280 KIAS, 26 degrees wing sweep, and 30 degrees bank in the turn.

INSTRUMENT DESCENT

1. Destination WX . . . . CHECK
2. Altimeter . . . . CHECK AND RECORD
3. Re-file . . . . AS NECESSARY
4. Penetration check list:
   a. Cabin air distribution lever (5 min. prior to descent) . . . . FWD DEFOG
   b. Cabin temperature . MAX. COMFORTABLE SETTING
   c. Probe heaters switch . . . . ON (AS NECESSARY)
   d. Fuel. . . . . DUMP AS NECESSARY
   e. Altimeter . . . . . SET PASSING FL 180

INSTRUMENT APPROACHES

Typical patterns on instrument approaches appear in figures 6-1 through 6-5. Adjust final approach speeds as necessary for a safe approach. Airspeed and configuration should be set up prior to descent on final. Do not begin lowering the gear or the slats/flaps above ___ KIAS and wing sweep 16 degrees. Lower the gear first, followed by the slats/flaps, allowing the airspeed to drop off to ___ KIAS after slats/flaps extension. The aircraft handles well through all normal speeds on the approach. If a missed approach, apply military thrust and retract the gear as soon as level-off is achieved and safe climb speed established. When the airspeed reaches 200 knots, retract the flaps. Continue the missed approach as directed, being careful not to exceed the missed approach altitude or to build up excessive airspeed. Establish a climb at 250 KIAS and limit turns to 30 degrees bank.

GCA APPROACHES

1. With gear and flaps down, fly pattern at units angle-of-attack (3:30 position on indicator).
2. Maintain ___ units angle-of-attack or "donut" airspeed (115 to 130 KIAS) on final approach. Approaching glide path, reduce power as necessary to maintain 500 to 700 rpm rate of descent.

ICE AND RAIN

ICING

Icing conditions should be avoided whenever possible. Before flight, check freezing levels and areas of probable icing from weather service. If ice starts to form on the windshield or wing leading edge, proceed as follows:

1. Probe heaters switch . . . . ON
2. Cabin air distribution lever . . . . FWD DEFOG

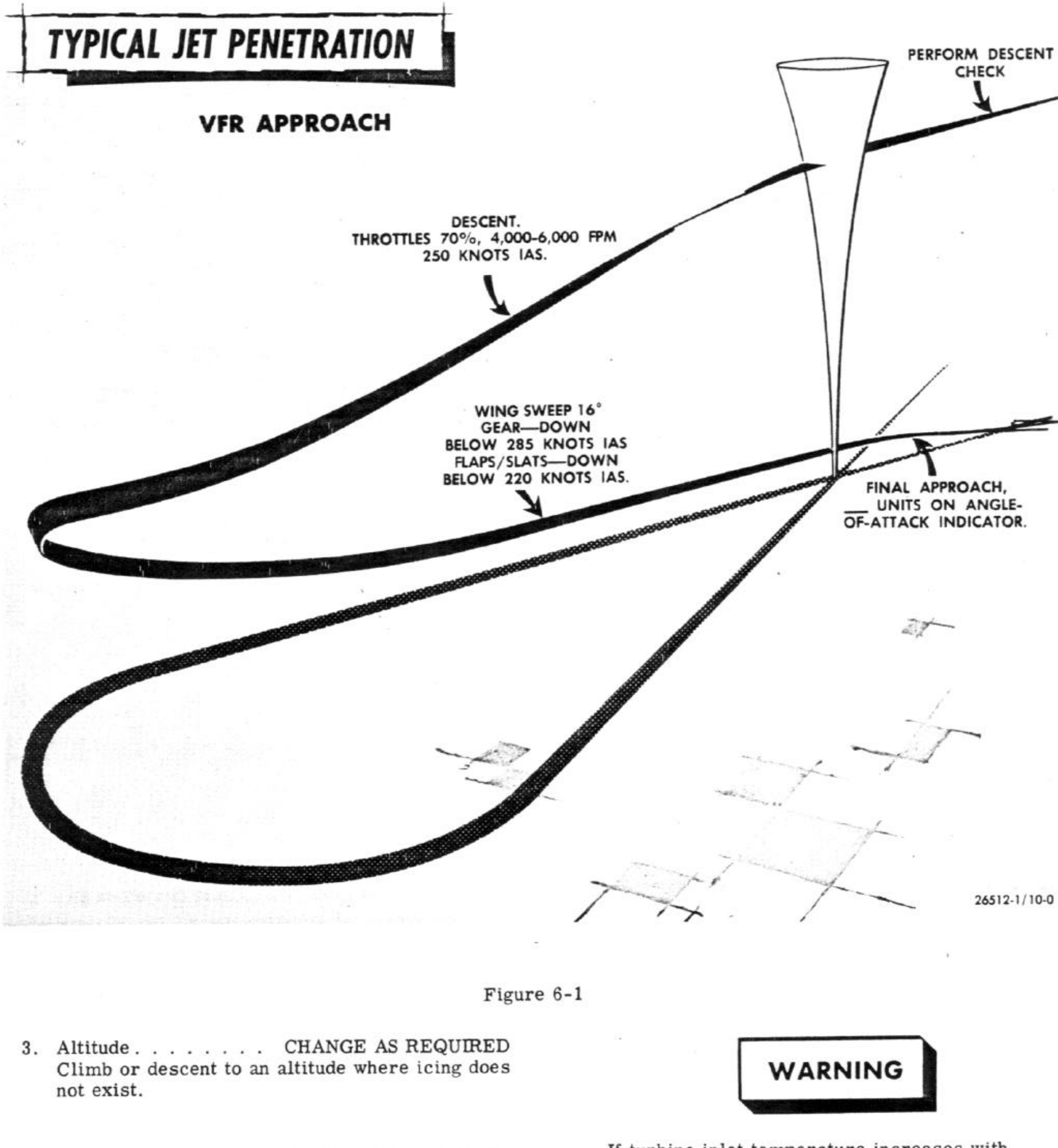


Figure 6-1

3. Altitude . . . . CHANGE AS REQUIRED
Climb or descent to an altitude where icing does not exist.

4. Engine instruments . . MONITOR FREQUENTLY
Carefully monitor tachometer and turbine inlet temperature indicator. A reduction of rpm or an increase in TIT accompanied by a loss of thurst is an indication of engine icing. If an indication of engine icing exists:

5. Engine anti-ice switch . . . . MAN

WARNING

If turbine inlet temperature increases with loss of thrust, the throttle should be retarded, as a low airspeed and high engine speed are conducive to engine icing.

A low approach into an area of moderate to severe icing should be considered an emergency approach. If there is any ice accumulation on the aircraft, an effort should be made to eliminate the ice before descending.

TYPICAL ADF APPROACH

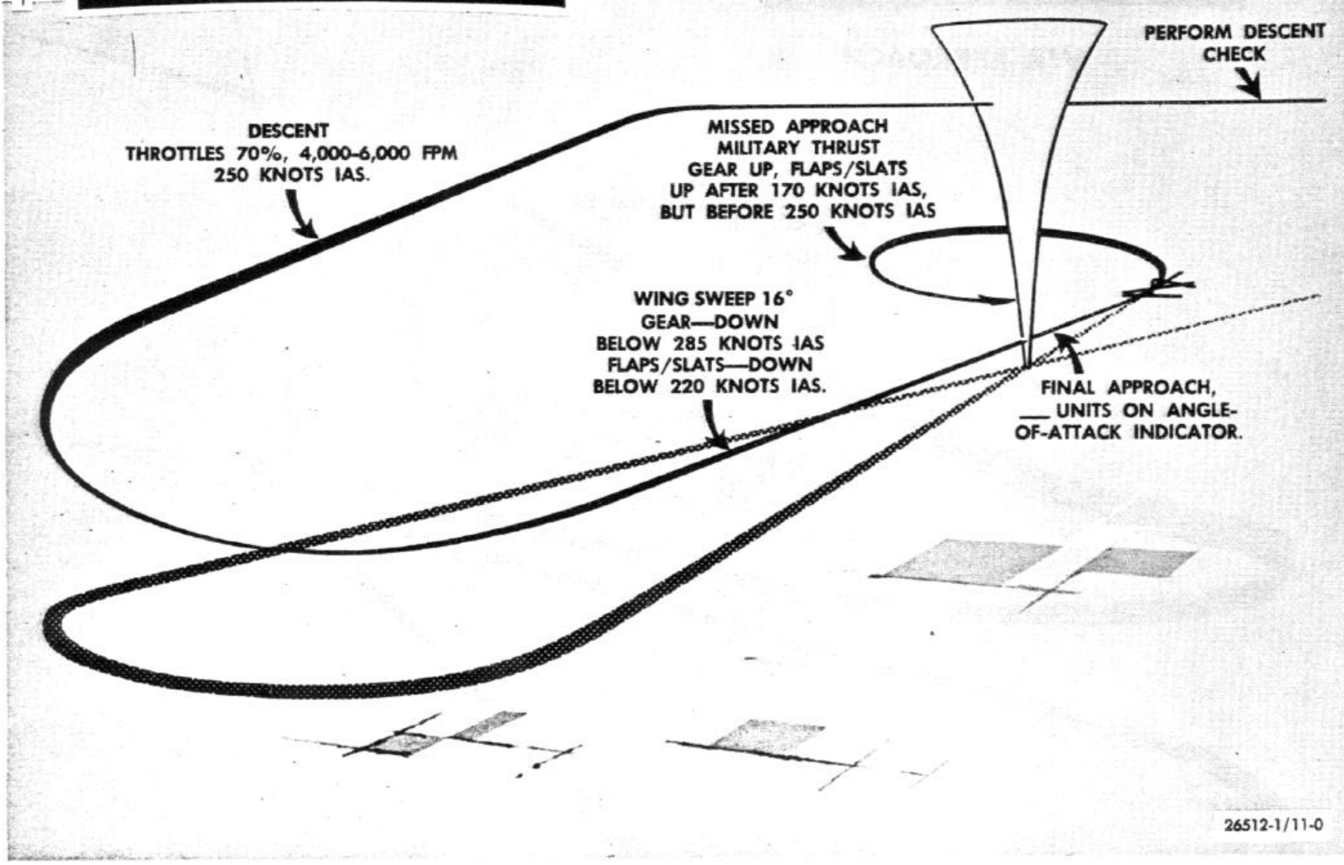


Figure 6-2

RAIN

Whenever rain is encountered, turn probe heaters switch ON.

Note

Under conditions of severe rainfall intensity, maintain a minimum engine power setting of 70% rpm. This will assure adequate acceleration margin and prevent possible engine speed "hang-up".

WINDSHIELD RAIN REMOVAL

CAUTION

Operation of the rain removal system is not recommended on a dry windshield. Use of the system on a dry windshield could result in:

1. Foreign materials being baked on the glass.
2. Above 360 KIAS, windshield distortion and failure.

LANDING IN RAIN

The RAIN REMOVAL position of the windshield wash/rain switch controls a blast of air that blows rain off the windshield. Be aware of the possibility of flame-out in a heavy rain and of reduced braking action due to a wet runway.

TURBULENCE AND THUNDERSTORMS

Intentional flight through thunderstorms should be avoided, unless the urgency of the mission precludes a deviation from course, due to the high probability of damage to the airframe and components by impact ice, hail, and lightning. Flame-outs, due to water ingestion, or compressor stalls due to rapidly changing flight attitudes, could also occur. The radar provides a means of navigation between, or around storm cells. If circumnavigating the storm is impossible, penetrate the thunderstorm in the lower third of the storm cell, away from the leading

TYPICAL TACAN APPROACH

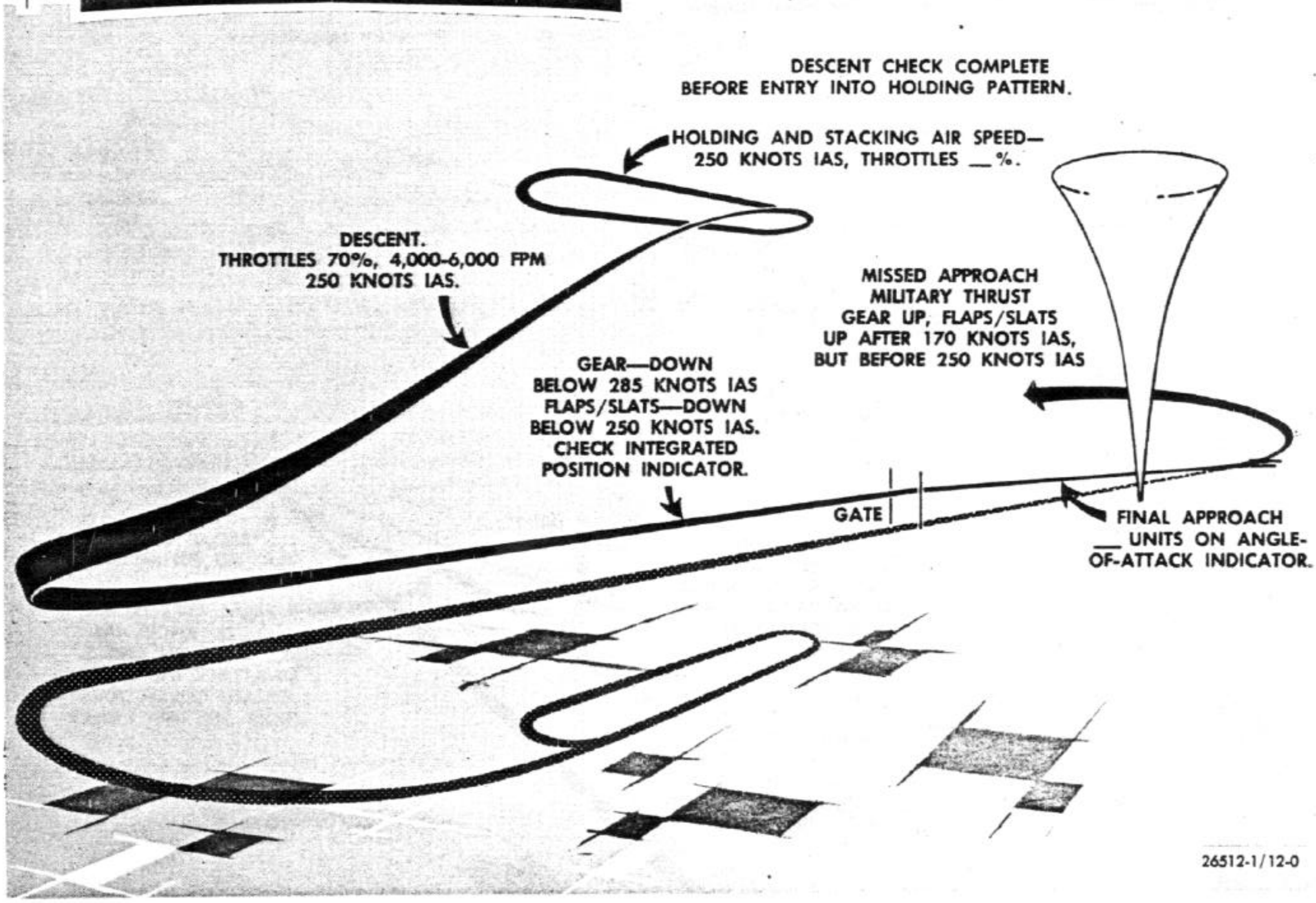


Figure 6-3

edge of the storm cloud, if possible. It is recommended that the NORM position of the AFCS be used. The AUTO mode should not be used. Structural damage could result with the automatic functions operating.

If necessary to penetrate a thunderstorm, proceed as follows:

1. Slow to between 275 to 300 KIAS.
2. Probe heaters switch . . . . .ON
3. Engine anti-ice switch (Recheck). . . . . AUTO
4. Loose equipment . . . . .SECURED
5. Tighten lap belt and lock should harness.
6. Cockpit lights . . . . .ON BRIGHT
7. Fly attitude and heading indicators primarily while in extreme turbulence, because altimeter and airspeed will fluctuate.

Note

During severe icing conditions, the pilot can expect to lose airspeed indications even with the probe heaters on. Ground radar stations, if available, can aid the pilot with tracking assistance through thunderstorm areas.

Severe turbulent air at high altitudes may cause the inlet airflow distribution to exceed acceptable limits of the engine, thereby inducing compressor stalls. To avoid compressor stalls during flight in turbulent air, maintain 275 to 300 KIAS at all altitudes.

Severe compressor stalls can result in engine flameout. If severe stalls are encountered, proceed as follows:

1. Throttles . . . . IDLE
2. Airspeed. . . INCREASE BY LOWERING NOSE

TYPICAL RADAR RECOVERY

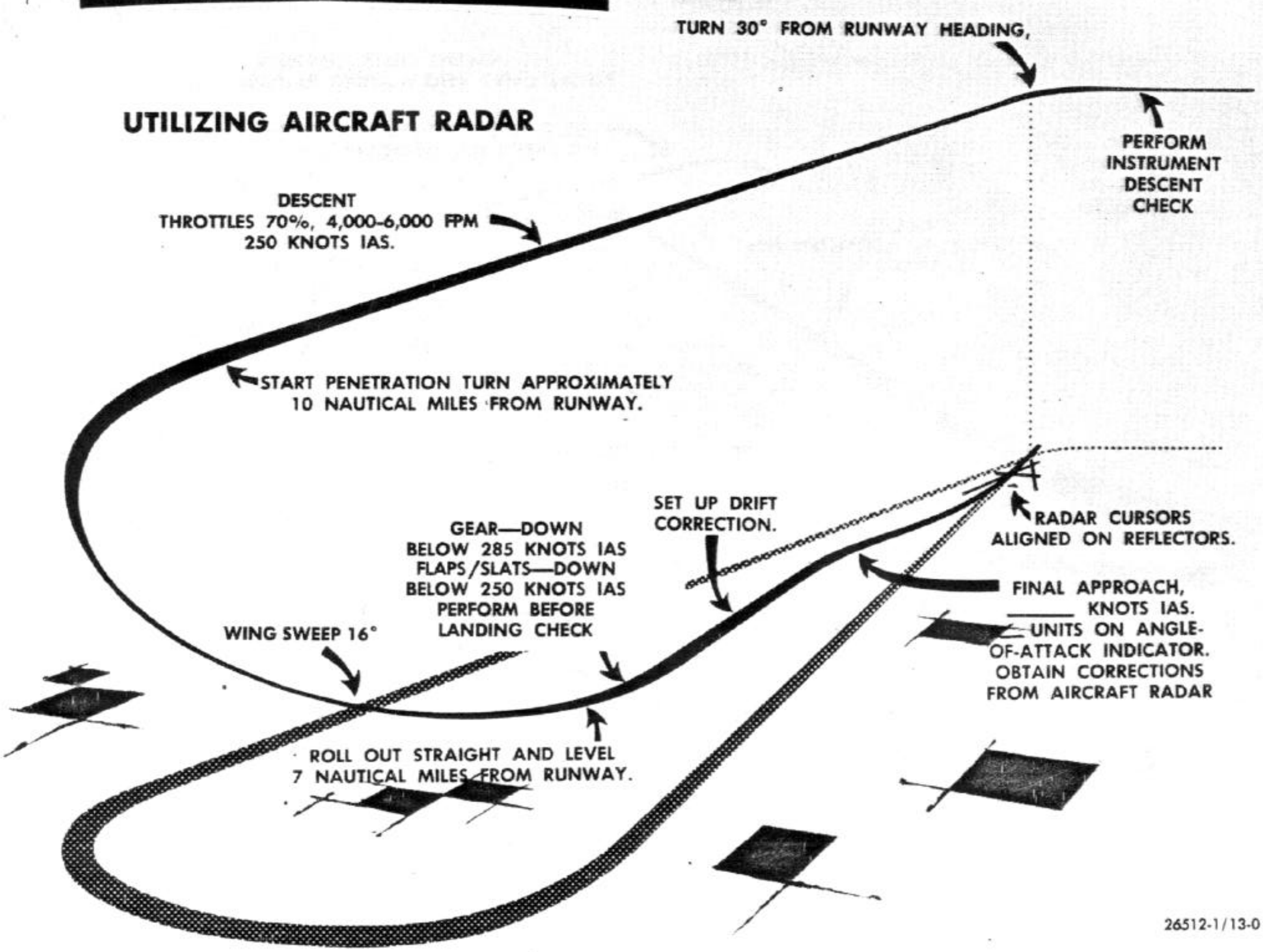


Figure 6-4

Compressor stalls are generally accompanied by increased TIT. If temperature exceeds allowable limits, perform shutdown procedure and accomplish an airstart as soon as practicable.

COLD WEATHER PROCEDURES

A careful pre-flight will eliminate many potential hazards found in cold weather operations. Inspect engine intakes for accumulation of ice and snow. If possible, pre-heat the engine for easier engine starts. When removing ice and snow from the aircraft surfaces, be careful not to damage the aircraft. Also, use precautions not to step on any no-step surfaces which could be covered with ice or snow. Check the pitot-static tube for ice as well as the fuel pressurization ram/air intakes, and yaw, pitch, and angle-of-attack transducers.

Moisture in the fuel system greatly increases operational problems in cold weather. At lower temperatures, the water-dissolving capacity of fuel is greatly reduced and will result in considerable more water accumulation (as much as several gallons of water to 1, 000 gallons of fuel). If the water separation occurs at below freezing temperatures, the water will crystallize on fuel drains and internal valves. Any water accumulation will settle to the bottom of the tanks and freeze up the fuel drains.

Normal operating procedures as outlined in Section III, Normal Procedures, should be adhered to with the following additions and exceptions:

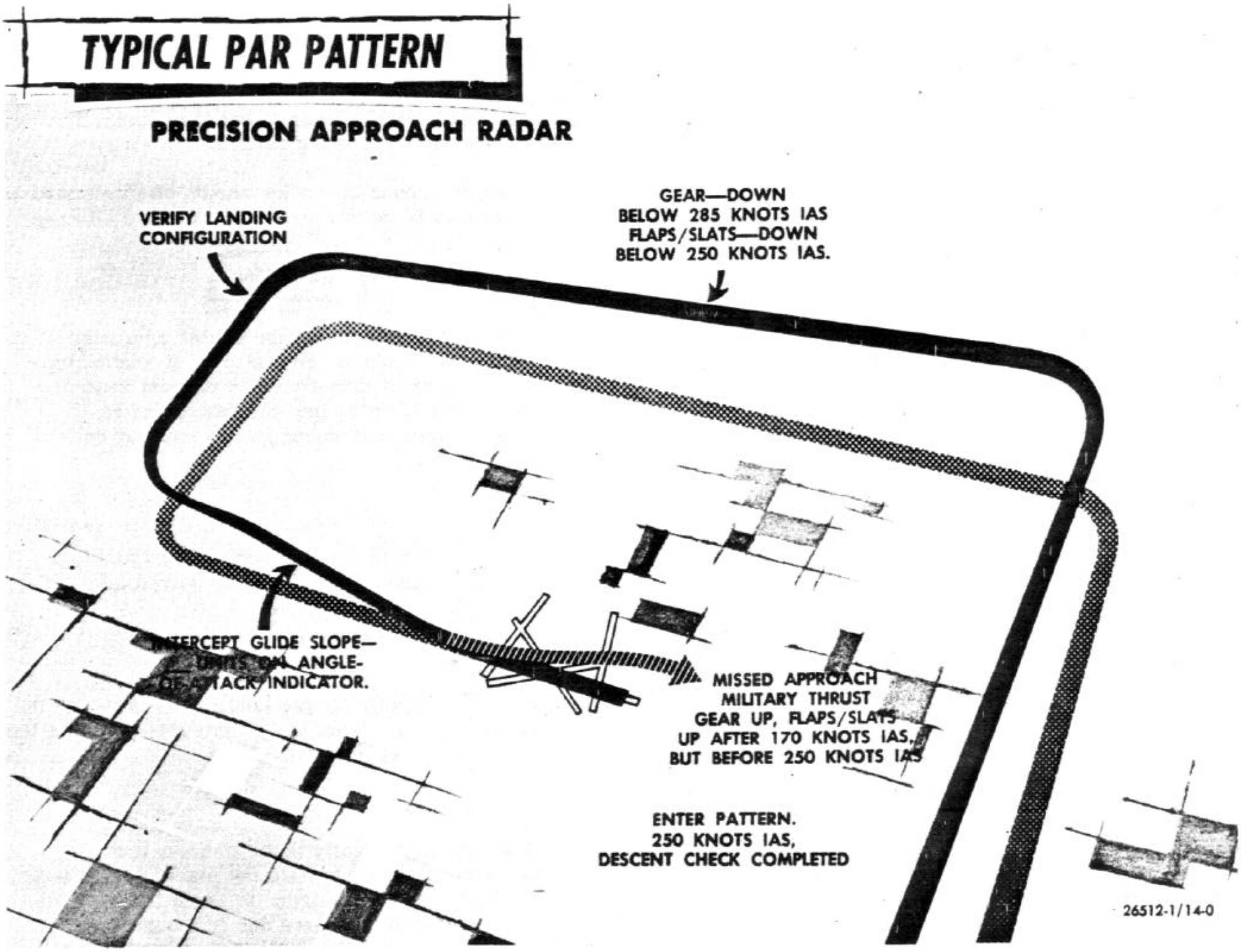


Figure 6-5

PRE-FLIGHT

1. Check entire aircraft to ensure that all snow, ice, or frost is removed.

WARNING

Snow, ice, and frost collections on the aircraft surface are a major flight hazard. The result of this condition is a loss of lift and increased stall speeds.

2. Shock struts and actuating cylinders . . . . FREE OF ICE AND DIRT

3. Fuel drain cocks . . . . . . FREE OF ICE AND DRAIN CONDENSATION

4. Pitot tube . . . . . . ICE AND DIRT REMOVED

5. Exterior protective covers . . . . . . REMOVED

WARMUP AND GROUND CHECK

Be sure that the aircraft is adequately checked before engine start. Normal starting procedures will start the engines in cold weather.

In severely cold weather, allow a short time for warmup before increasing RPM out of the idle range. If oil pressure is low or fails to come up in a reasonable length of time, shut down. Attempt another start after heating the engines.

WARNING

If abnormal sounds or noises are present during starting, discontinue starting and apply intake duct pre-heating for 10 to 15 minutes.

TAXIING

1. Avoid taxiing in deep or rutted snow since frozen brakes will likely result.

2. To ensure safe stopping distance, and prevent icing of aircraft surfaces by melted snow and ice blown by jet blast of a preceding aircraft, increase spacing between aircraft while taxiing at sub-freezing temperatures.

TAKEOFF

Thrust available will be noticeably greater in cold temperatures during the take-off run.

CAUTION

Prior to initial take-off roll, ensure that all instruments are sufficiently warmed up. After takeoff, cycle landing gear a few times to free the gear from the possibility of freezing in the wheel wells.

LANDING

Use anti-skid during the landing roll.

Note

Hard braking on ice or wet runways, even with anti-skid system on could result in dangerous skidding conditions.

HOT WEATHER AND DESERT PROCEDURES

Check for accumulation of sand or dust in the intakes and transducers. Normal starting procedures will be employed.

Normal operating procedures as outlined in Section III, Normal Procedures, should be adhered to with the following additions and exceptions:

1. Expect higher temperatures than normally obtained in operating ranges.

2. Engine ground operation should be minimized as much as possible.

CAUTION

Do not attempt takeoff or engine operation in a sand storm or dust storm, if avoidable. Park aircraft cross wind to prevent sand or dirt from blowing into the intake and exhaust ducts, and subsequently causing engine damage.

TAKEOFF

The take-off distances are increased by ambient temperature increases. Check required take-off distance charts in Section XI.

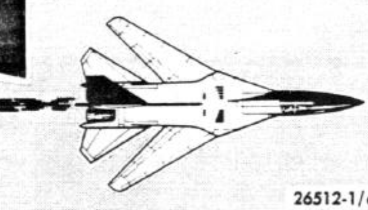
LANDING

Anticipate a slightly longer landing distance and the possibility of turbulence due to thermal action of the air close to the ground.

SHUTDOWN.

Open the canopy slightly if the weather and environment permits it. Do not place objects near the cockpit windows in order to avoid the possibility of cracking the windows due to a concentration of radiant energy. Check all protective covers installed.

section VII
COMMUNICATIONS EQUIPMENT AND PROCEDURES



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COMMUNICATIONS AND ASSOCIATED ELECTRONICS EQUIPMENT

See figure 7-1 for communications and associated electronic equipment.

INTERCOMMUNICATIONS SET (ICS) AN/AIC-25

The intercommunications set (ICS) provides communications between crewmembers and crewmembers and ground crew. The ICS also provides the necessary amplification for UHF transmission and reception, TACAN identification, ECM tone reception and BOMB tone transmission and reception. Two identical ICS control panels (figure 7-2) on the left and right consoles are provided for the pilot and MCO. Interphone stations for ground crew operation are in the nose wheel well and aft equipment bay. Power is supplied to the ICS from the 28-volt essential DC bus.

COMMUNICATIONS MONITOR SELECTOR

Eight communication monitor selectors are on each ICS panel. Only six monitor selectors are presently used and monitor the following functions:

INPH	-	Interphone
UHF/AUX	-	UHF Communications Set AN/ARC-51B (MAIN) UHF Auxiliary Receiver AN/ARR-69 (AUX)
TACAN	-	TACAN Identification
ECM	-	ECM Tone
BOMB	-	BOMB Tone
UHF #2	-	UHF Communications Set AN/ARC-51A
IR	-	Not Operational

The monitor selectors are pulled out to turn on and pushed in to turn off. When pulled out, each selector may be rotated to the desired volume level.

MASTER VOLUME CONTROL

A master volume control on each ICS panel controls the volume of all inputs to the panel. To change an individual input, it can be accomplished by rotating the appropriate monitor selector to the desired level.

HOT MICROPHONE BUTTONS

A push-pull (HOT MIC) hot microphone button on each ICS control panel provides a continually operating microphone when pulled out. This permits a crewmember to transmit on interphone without manual ICS keying.

CALL BUTTON

The CALL button is on the ICS control panel. Depressing either CALL button boosts the interphone volume of the other stations and reduces the operator's side tone level, allowing the call signal to override all inputs to the other ICS stations.

TRANSMITTER SELECTOR SWITCHES

Two seven positioned transmitter selector switches are on the ICS control panel. Only four positions of the selector switches are currently functional and are labeled INT, UHF, UHF #2, and TONE. The UHF position selects the AN/ARC-51B transmitter

COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT

TYPE AND DESIGNATION	FUNCTION	RANGE	OPERATOR	LOCATION OF CONTROLS
INTERPHONE AN/AIC-25	VOICE COMMUNICATIONS BETWEEN CREW OR CREW AND GROUND PERSONNEL. RADIO SELECTION. WARNING TONES FROM WEAPONS AND ECM.	WITHIN THE AIRCRAFT AND GROUND CREW PERSONNEL.	PILOT, MCO, GROUND CREW PERSONNEL.	PILOT'S SIDE CONSOLE, MCO'S SIDE CONSOLE, NOSE WHEEL WELL AND AFT EQUIPMENT BAY
UHF COMMUNICATIONS SET AN/ARC-51B	TWO-WAY VOICE COMMUNICATIONS.	LINE-OF-SIGHT.	PILOT, MCO	CENTER INSTRUMENT PANEL.
UHF AUXILIARY RECEIVER AN/ARR-69	AUXILIARY UHF RECEIVER	LINE-OF-SIGHT.	PILOT, MCO	CENTER INSTRUMENT PANEL.
UHF DIRECTION FINDER AN/ARA-50	PROVIDES BEARING INFORMATION TO SELECTED UHF STATIONS.	LINE-OF-SIGHT.	PILOT, MCO	CENTER INSTRUMENT PANEL.
UHF COMMUNICATIONS SET AN/ARC-51A(UHF 2)	TWO-WAY VOICE COMMUNICATIONS.	LINE-OF-SIGHT.	MCO	MCO'S SIDE CONSOLE.
TACAN AN/ARN-52	PROVIDES BEARING AND DISTANCE INFORMATION.	LINE-OF-SIGHT UP TO 300 MILES DEPENDING ON ALTITUDE.	PILOT, MCO	CENTER INSTRUMENT PANEL.
IFF TRANSPONDER AN/APX-64(V)	AIRCRAFT IDENTIFICATION.	LINE-OF-SIGHT.	PILOT, MCO	CENTER CONSOLE.
RADAR ALTIMETER AN/APN-167	INDICATES DISTANCE IN FEET FROM AIRCRAFT TO SURFACE.	0-5000 FEET	PILOT	PILOT'S INSTRUMENT PANEL.

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Figure 7-1.

PILOT'S AND MCO'S RADIO AND ICS CONTROLS

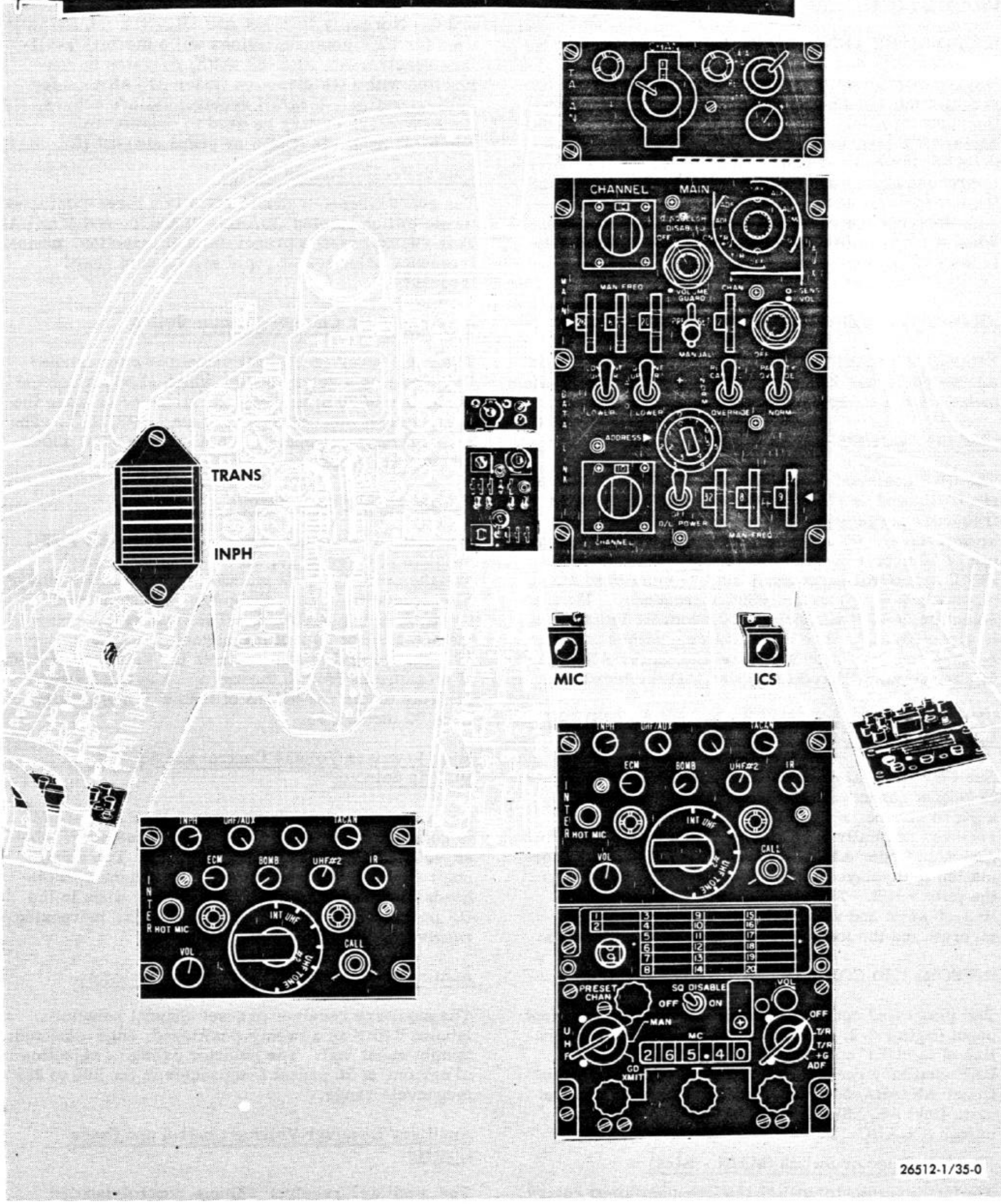


Figure 7-2.

and the UHF #2 selects the AN/ARC-51A transmitter. Only the transmitter selected will be keyed when a microphone switch is actuated. The TONE position enables a bomb tone signal to be transmitted over the UHF mike line.

MICROPHONE AND ICS SWITCHES

A three-positioned, pivot-type microphone switch for the pilot labeled TRANS and INPH (figure 7-2) is on the right throttle. The switch is spring-loaded to the center OFF position. The switch is moved forward (TRANS) for radio transmissions or aft for (INPH) interphone operation. A microphone switch (MIC) on the left footrest and an interphone switch (ICS) on the right footrest are provided for the MCO (figure 7-2). When the transmitter selector switch on ICS panel is in the INT position, keying either the TRANS (MIC) or the INPH (ICS) switch allows interphone use.

EXTERIOR INTERPHONE STATIONS

Exterior interphone stations in the nose wheel and the aft equipment bay have a volume control, a call push-button, and a receptacle for ground cord plug in.

UHF COMMUNICATIONS SET AN/ARC-51B

The UHF communications set provides air-to-air and air-to-ground or shipboard communications. The frequency range extends from 225.00 to 399.95 mc (megacycles). The equipment allows selection of any one of 20 preset channels including guard channel of 243.0 mc. Guard frequency may be monitored simultaneously with any other selected frequency. Manual selection of 3500 channels in 50 kilocycle increments is also provided. The UHF communications set may be used in conjunction with direction finder AN/ARA-50 for automatic direction finder (ADF) operation.

UHF AUXILIARY RECEIVER AN/ARR-69 AND UHF DIRECTION FINDER AN/ARA-50

The UHF auxiliary receiver provides for reception of 19 preset channels (265 to 285 megacycles) and a guard channel of 243.0 mc. The UHF auxiliary receiver normally operates in conjunction with UHF direction finder AN/ARA-50. Relative bearing information is displayed by the NO. 2 bearing pointer on the pilot's HSI. The operating range is limited to line-of-sight and will vary with the altitude of the aircraft and the location of the transmitting station.

INTEGRATED CONTROL PANEL

The integrated control panel on the center instrument panel (figure 7-2) contains all the controls for operation of the UHF communications set AN/ARC-51B, UHF auxiliary receiver AN/ARR-69, UHF direction finder AN/ARA-50, digital data communications set (data link) AN/ASW-27, and UHF data link transceiver AN/ARC-124.

Function Selector Switch (MAIN - AUX)

The function selector switch is a five-positioned rotary switch on the integrated control panel. The switch positions are arranged to provide complementary functions of the UHF communications set AN/ARC-51B (MAIN) and the UHF auxiliary receiver AN/ARR-69 (AUX). The MAIN portion of the switch is labeled OFF, T/R, T/R+G, ADF, and ADF, while the corresponding AUX portion is labeled ADF, ADF, CMD and G. Normally UHF set AN/ARC-51B (MAIN) is used for UHF communications while the UHF auxiliary receiver AN/ARR-69 (AUX) operates in conjunction with UHF direction finder AN/ARA-50 for ADF operation. If MAIN receiver failure occurs, the AUX receiver may be used to monitor voice command channels (CMD) or guard channel (G).

Guard-Preset-Manual Switch

The guard-preset-manual switch is a three-positioned toggle switch labeled GUARD, PRESET, and MANUAL. This switch permits preset channel selection, manual frequency selection or rapid selection of guard frequency.

Main Receiver Channel Selector Switch

The main receiver channel selector switch labeled CHANNEL is a rotary switch which allows selection of any one of 20 preset channel frequencies when the guard-preset-manual switch is set to PRESET. The selected channel appears in a display window above the selector switch.

Manual Frequency Controls

The manual frequency controls labeled MAN FREQ consists of three-edge, mounted, thumb-wheel switches which permit manual selection of any one of 3500 frequencies when the guard-preset-manual switch is set to MANUAL. The first switch controls the selection of 10's of megacycles from 22 through 39. The second switch controls the selection of units of megacycles from 0 through 9. The third switch controls tenths and halves of tenths of megacycles from 00 through 95.

Main Receiver Volume Control and Squelch Disable Selector

The main receiver volume control labeled VOLUME is concentric with the squelch disable selector labeled SQUELCH DISABLED (OFF - ON). The volume control adjusts the audio level of the signals to the headset. The squelch disable selector when in the ON position disables the squelch circuit, increasing receiver sensitivity.

Auxiliary Receiver Pre-Set Channel Selector

The auxiliary receiver pre-set channel selector labeled CHAN is a twenty-positioned, edge-mounted, thumb-wheel dial. The selector provides selection of any one of 20 preset frequencies in the 265 to 285 megacycle range.

Auxiliary Receiver Volume Control and Sense Control

The auxiliary receiver volume control labeled VOL is concentric with the sense control labeled SENS. The volume control adjusts the audio level

of the signals delivered to the headset. The sense control increases or decreases receiver sensitivity.

Communications Antenna Reversing Switch

The communications antenna reversing switch is a three-positioned toggle switch labeled UPPER, AUTO, and LOWER. Selection of AUTO allows the system to select the antenna which provides the strongest audio signal to the headset. The UPPER or LOWER position provides for selection of either top or bottom UHF communications antennas.

UHF COMMUNICATIONS SET AN/ARC-51A (UHF-2)

The UHF communications set AN/ARC-51A provides voice communications between aircraft and ground or shipboard stations. The frequency range extends from 225.00 to 399.95 megacycles (mc). The equipment allows selection of any one of 20 preset frequencies including guard channel of 243.0 mc. Guard frequency may be monitored simultaneously with any other selected frequency. Manual selection of 3500 channels in 50 kilocycle increments is also provided. The ADF mode is not operational.

UHF COMMUNICATIONS CONTROL PANEL (UHF 2)

The UHF communications control panel (figure 7-2) is on the MCO's side console. A four-positioned function selector switch turns the equipment ON when advanced from the OFF position. In the T/R position, only the main receiver-transmitter is monitored; in the T/R & G position, both the main receiver and guard receiver are monitored. The ADF position of the function selector switch is inoperative. A three-positioned mode selector switch when in the PRESET CHAN position allows the selection of any one of the 20 preset channels. The channel is selected by rotating the preset channel control knob to the desired channel shown on the preset channel indicator. When in the MAN position, any one of 3500 possible frequencies may be selected manually by setting the three frequency selectors to the desired frequency. When the mode selector is in the GD XMIT position and the function selector switch is in the T/R position, the guard channel is selected for UHF transmission and reception. With the function selector switch in the T/R & G position, the guard receiver is also turned on providing two receivers for guard reception. The VOL control adjusts the audio level of the signals delivered to the headset. A two-positioned toggle switch labeled SQ DISABLE (OFF-ON) when in the ON position disables the squelch circuit, increasing receiver sensitivity.

TACAN NAVIGATIONAL SET AN/ARN-52 (V)

The TACAN navigational set enables the aircraft to receive continuous indications of its distance and bearing from any selected TACAN station located within a line-of-sight distance of approximately 300 nautical miles. Station identification is also provided through the ICS system. There are 126 channels available for selection. An air-to-air mode is available which can be used between two cooperating aircraft having TACAN with air-to-air capability, for range information only. Bearing, course deviation, distance and to-from information to the selected TACAN station is displayed on the pilot's HSI. The TACAN set operates on 28-volt DC from the essential DC bus and 115-volt AC from the essential AC bus. The TACAN control panel is on the center instrument panel (figure 7-2).

TACAN FUNCTION SELECTOR SWITCH

The function selector switch on the TACAN control panel has four positions labeled OFF, REC, T/R, and A/A. When the selector switch is set to the REC (receive) position, only bearing information is available. In the T/R (transmit/receive) position, both range and bearing information is available. When A/A (air-to-air) operation is desired, a 63 channel difference must be selected between cooperating aircraft. Both aircraft must select A/A. In the A/A mode, the TACAN will provide range between aircraft only (no identity or bearing).

TACAN CHANNEL SELECTOR

The channel selector on the TACAN control panel consists of inner and outer adjustment controls for selecting any one of the available 126 TACAN channels. The outer control is used to select the first two digits of the desired channel and the inner control to select the last digit.

TACAN VOLUME CONTROL SELECTOR

A volume control selector on the TACAN control provides a means of regulating the volume of the identity tone to the ICS.

IFF/SIF TRANSPONDER SET AN/APX-64 (V)

The IFF/SIF transponder set provides the aircraft with an automatic means of selective identification. The system replies to proper IFF interrogation. Mode 2 code settings are set into the receiver-transmitter on the ground and thus are fixed for any one flight. Mode 1 and mode 3/A codes are set up at the control panel. The system provides identification of position and emergency replies. An optional setting (LOW) reduces receiver sensitivity so that replies are made only to the geographically nearest interrogators. Electrical power is supplied to the IFF system from the 115-volt essential AC bus and 28-volt essential DC bus.

IFF/SIF CONTROL PANEL

The IFF/SIF control panel (figure 7-3 is on the center console. The panel contains the master control switch, identification-of-position switch, four mode select/test switches and six thumb actuated code selector switches. MODE 4, MODE C, and the RAD TEST-MON switch are not currently operational.

IFF Master Control Switch

The IFF master control switch has OFF, STBY, LOW, NORM, and EMER positions. When in the STBY position, the equipment is warmed up but will

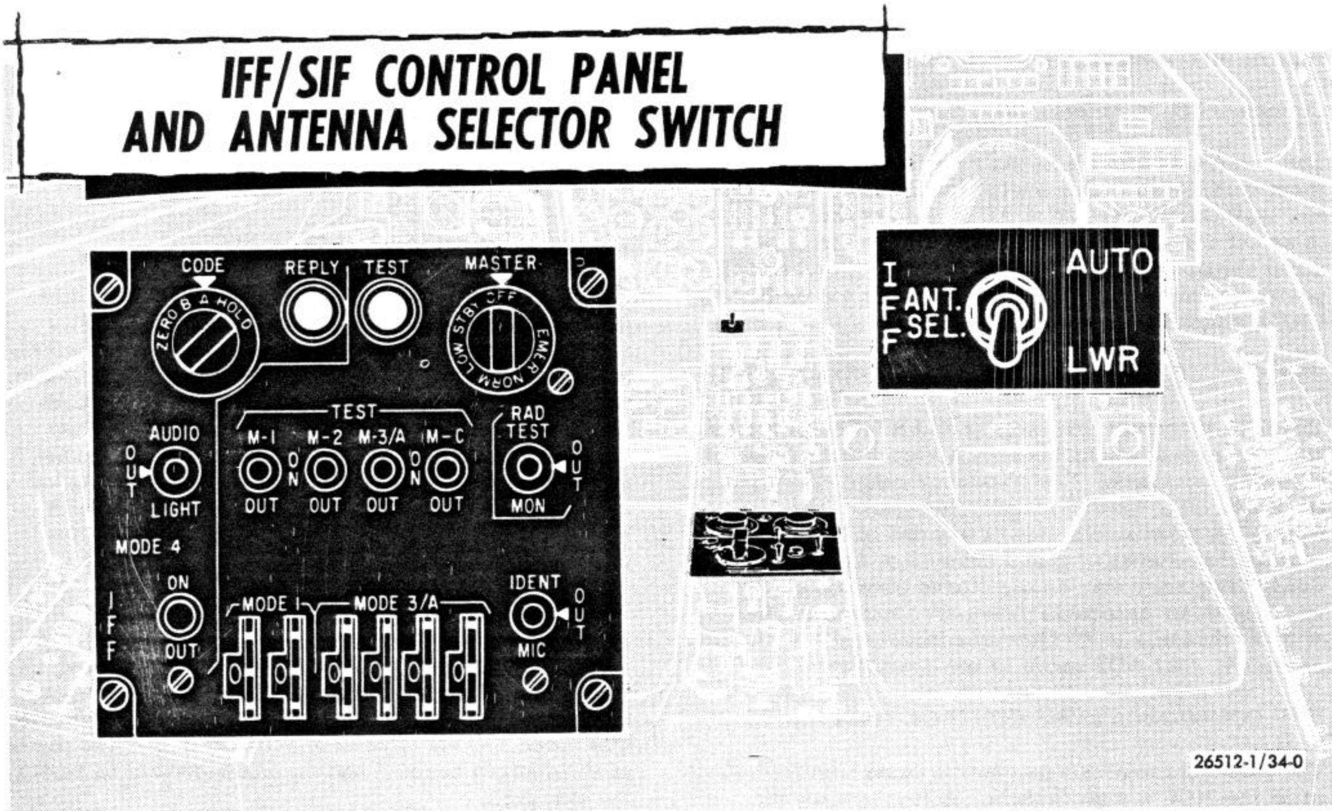


Figure 7-3.

not receive or transmit. With the control set at LOW, the system operates at reduced receiver sensitivity on all modes and requires a stronger (geographically closest) interrogation. When positioned to NORM, full range operation and reply occurs. Transmitted power is the same for both the LOW and NORM positions. To select the EMER position, the control must be pulled outward. In the EMER position, an emergency reply is transmitted each time a mode 1, mode 2 or mode 3/A interrogation is recognized.

Mode Selector/Test Switches

Four mode selector/test switches labeled M-1, M-2, M-3/A and M-C have TEST, ON and OUT positions. In the TEST position, the TEST light will illuminate, indicating proper operation of the mode selected. The OUT position for each switch disables its respective mode. The ON position for each switch enables the transmitter-receiver to reply to interrogations for the mode selected. The mode C selector/test switch is not currently operational.

Code Selector Switches

Six code selector switches are provided for selection of mode 1 and mode 3 codes. Mode 1 has two thumbwheel selectors which allow selection of 32 different codes. Mode 3 has four thumb actuated selector switches which provide the capability of selecting 4096 codes.

Identification and Position Switch

The identification and position (I/P) switch has three positions labeled IDENT, OUT, and MIC. When the switch is momentarily held in the spring-loaded IDENT position, the I/P timer is energized for 30 seconds. If a mode 1 or mode 3 interrogation is recognized within this 30-second period, I/P replies will be transmitted. When the switch is placed in the MIC position, an I/P pulse group will be transmitted in reply to a mode 1 or mode 3 interrogation as long as the UHF transmitter is keyed and for 30 seconds afterwards. The OUT position prevents transmission of I/P replies.

IFF Antenna Selector Switch

The two-positioned IFF antenna selector switch on the forward portion of the center console (figure 7-3) is labeled AUTO and LWR. When the switch is placed to AUTO, the antenna lobing switch cycles contact of the receiver-transmitter between the upper and lower antenna to provide relatively complete antenna pattern coverage. When the antenna selector switch is placed to LWR, the lower antenna will be used to receive and reply to interrogation signals below the aircraft. The upper antenna radiation pattern has a slight forward tilt, and the lower antenna radiation pattern has a slight aft tilt.

RADAR ALTIMETER SYSTEM AN/APN-167

The radar altimeter is a low altitude, pulsed, range-tracking radar that measures the surface or terrain clearance below the aircraft in the range of 10 to 5000 feet. Altitude information is derived by radiating a short duration RF pulse from the transmit antenna to the earth's surface, and measuring the time interval until the RF energy returns through the receive antenna to the receiver. The altitude information is continuously presented to the pilot, in feet of altitude, on the dial of an indicator (figure 7-4) on the pilot's instrument panel.

The radar altimeter has two modes of operation, search and track. In the search mode, the system successively examines increments of range until the complete altitude range is scanned for a return signal. When a return signal is detected, the system switches to the track mode and tracks the signal, giving continuous altitude information.

The system provides reliable altitude information in a range of 10 to 5000 feet and permits close altitude control at minimum altitudes. It has an accuracy to within 5 feet ±3 percent of actual altitude. The system is inoperative in banks of more than 30°, in climbs or dives of more than 50°, and when the reflected signal is otherwise too weak.

Radar altitude information is also supplied to the VDIG, AMCS and the data link system through the analog-to-analog converter. On the extreme right side of the PI, an altitude scale from 0 to 1500 feet and a movable pointer is present in the TAKEOFF and LANDING modes.

The radar altimeter operates on 28-volt DC from the essential DC bus and 115-volt AC from the essenti?' AC bus.

RADAR ALTIMETER INDICATOR

The altitude indicator (figure 7-4) is in the upper right-hand corner of the pilot's instrument panel and contains the only operating controls in the system. Altitude is displayed on a single-turn dial that is calibrated from 0 to 5000 feet in increasing increments of 10 feet up to 500 feet, 50 feet from 500 up to 1000 feet, and 500 feet from 1000 up to 5000 feet. This permits greater definition at lower altitudes.

ON/OFF/Self-Test Control

The radar altimeter control knob on the lower right of the indicator is a combination power switch, self-test button, and an altitude limit index selector. Initially turning the knob clockwise energizes the system. Further rotating the knob positions the altitude index pointer to a desired setting for reference in flying minimum altitudes. Depressing and holding the knob activates the self-test circuit; presented on the indicator is a synthetic target at 100 feet ±10

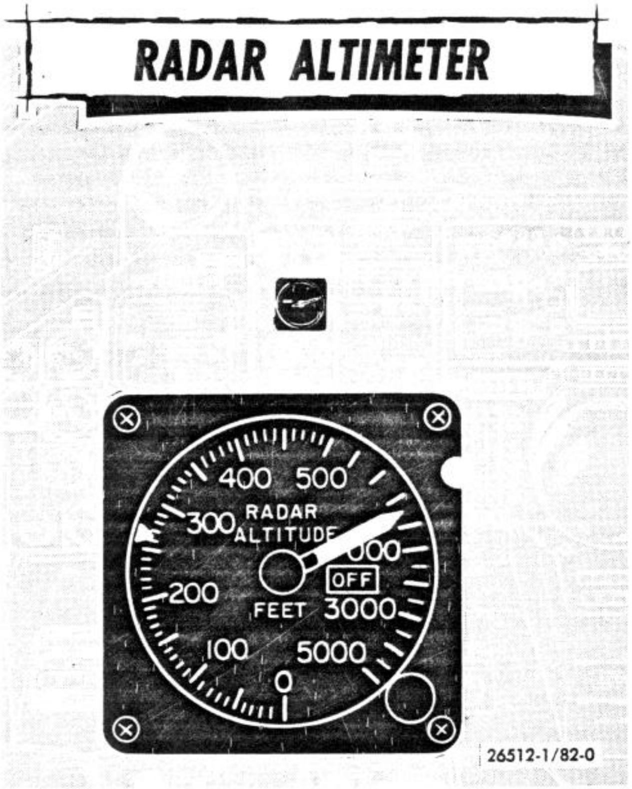


Figure 7-4

feet. This test feature is available at altitudes below 38,000 feet. Normal operation is resumed by releasing the control knob.

OFF Flag

The OFF flag in the middle of the indicator is actuated either electronically or mechanically. It is energized electronically when the altitude signal becomes unreliable and exceeds a threshold voltage. Mechanical actuation of the OFF flag occurs when the altitude pointer rotates beyond 5,000 feet and actuates the OFF flag switch.

Low Altitude Warning Lamp

A low altitude warning lamp is on the front plate of the DVI of the VDIG. Whenever aircraft altitude falls below the pilot's preset limit index, the RADAR ALT LOW lamp will illuminate.

Unreliable Radar Altitude Signal

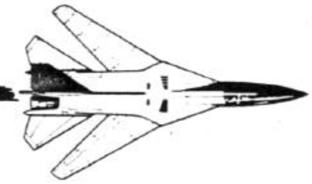
The radar altimeter is tied to the central air data computer through the analog-to-analog converter which will repeat radar altitude. This permits incremental barometric altitude changes from the last valid radar information to be substituted for radar altitude whenever the system malfunctions or the radar signal is unreliable.

During periods of radar unreliability, the CADC will supply the subsystems with barometric altitude changes until a reliable radar signal is received. Because of this tie-in with the CADC, the altitude displayed on the height indicator may not always be an accurate indication of elevation above the earth's surface. When the radar signal is unreliable, the OFF flag will appear on the indicator and is the only indication the pilot has that the radar altimeter system is not providing subsystems altitude information.

Note

The appearance of the OFF flag is the only indication that actual radar altitude above the terrain is no longer being displayed.

section VIII
WEAPONS SYSTEM



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INERTIAL NAVIGATION SYSTEM AN/AJN-14 (INS)

The inertial navigation system (INS) is a self-contained automatic aid to navigation, independent of outside reference. Three basic units comprise the system: the inertial reference, the navigation computer and the navigation control panel.

The inertial reference unit provides aircraft heading, pitch and roll attitude, and horizontal and vertical velocities. Position, course and distance to destination computations are performed by the navigation computer, using data from the inertial platform. The navigation control panel provides switches, selectors and indicators for monitoring, controlling and testing the INS. System accuracy is within 2 nautical miles of the indicated position in any one-hour period.

INERTIAL REFERENCE UNIT

The inertial reference unit is comprised of an inertial platform and an analog inertial computer. Inertial platform orientation is maintained with respect to three mutually perpendicular axes referred to as X, Y, and Z. The Z-axis is coincidental with the vertical, and the direction toward the center of the earth is considered to be positive. The X-axis, perpendicular to the Z-axis, lies in the plane of the local meridian, and its positive direction defines north. The Y-axis, mutually perpendicular to the X-axis and Z-axis, is positive in the direction of earth's rotation, which is east.

The platform is mounted in a four gimbal structure. This allows the aircraft freedom of motion while the platform is permitted to retain its orientation regardless of aircraft maneuvers. Accelerometers measure aircraft acceleration about the X, Y, and Z axes. These inertially-sensed accelerations are corrected for coriolis and then integrated in the inertial computer to produce velocity. The inertial computer returns signals corrected for aircraft velocity and earth-rate to the platform gyro torques to maintain the platform level with respect to gravity and oriented to true north. Synchros mounted on the platform gimbals generate attitude and azimuth signals which provide pitch, roll, and true heading data to the navigation computer. Velocity information is also transmitted from the inertial computer to the navigation computer where it is integrated to produce aircraft position data.

NAVIGATION COMPUTER

The navigation computer is an airborne analog dead reckoning computer which continuously and automatically computes and displays aircraft present position in latitude and longitude. It provides for selection, storage and display of latitude and longitude of three destination positions; computes range and course angle to destination, wind magnitude and direction, groundspeed and groundtrack and aircraft steering information. If range is less than 200 nautical miles, SR NAV is selected and rhumb line computations are used in the computer. For distances over 200 nautical miles, GC NAV is selected and great circle computations are used. Present position may be corrected manually or automatically by overfly fix methods or by radar cursor positioning through the search radar. The computer contains self-test capabilities for ground checks of computer reliability. Computer inputs are received from the air data computer, inertial platform, search radar and the flux valve. Outputs are sent to the inertial platform, vertical display indicator group, airborne missile control, automatic flight control system, search radar, and flight instruments. The computer operates in the normal navigation mode unless the inertial platform is inoperative, then it will automatically change over to an auxiliary navigation mode.

ALIGNMENT

In order to establish the inertial navigation system as a valid reference, the inertial platform must be leveled in relation to local gravity and oriented to true north. This procedure is known as alignment and is accomplished in two phases: coarse alignment and fine alignment.

Coarse alignment begins as the gyros are brought up to speed and the platform is leveled with respect to

the pitch and roll axes of the aircraft, which will normally be no more than 10 degrees off the earth's tangent plane. Leveling action is accomplished by a null-seeking servo loop; error signals from the gimbal synchro transmitters are transmitted to platform gimbal motors until a null is achieved on the transmitters. The azimuth gimbal is coarsely aligned, using magnetic heading information from the flux valve and magnetic variation manually inserted to derive true heading. This true heading is compared to the actual azimuth angular position of the platform. The difference (error signal) is transmitted to gimbal motors to reposition the azimuth axis until a null is achieved.

Fine alignment, which begins automatically at termination of coarse alignment, consists of two parts: leveling and gyrocompassing. Leveling consists of accurately stabilizing the gyro platform. When leveling is entered, the accelerometers sense any displacement of platform orientation with respect to gravity. Accelerometer outputs are processed in the inertial computer to generate gyro torquing signals. Gyro output signals generated are applied to platform gimbal motors to bring accelerometer outputs to zero. When the accelerometer outputs are stabilized, they are near a null and the platform X and Y axes are perpendicular to the earth's gravity (local vertical).

After approximately one minute of leveling (carrier alignment requires four minutes), a fine degree of leveling is obtained and the alignment procedure automatically goes into gyrocompassing. This is a method of accurately orienting the platform X-axis to true north. The X (North) accelerometer outputs are at a null when the Y (East) gyro input axis senses zero earth rotation. These X accelerometer outputs are processed in the inertial computer and used to align the platform. When a stabilized condition is achieved, the X-axis is aligned perpendicular to the direction of earth's rotation (East), the Y-axis is aligned with true north, and the vertical axis is aligned with the center of the earth. A minimum of four minutes is required by the inertial computer for the gyrocompassing mode during ground alignment.

Ground Alignment

Ground alignment of the inertial platform consists of accomplishing coarse and fine alignment previously discussed.

The inputs required to align the inertial platform are present position and true heading. After ALIGN mode is entered, present position is inserted by slewing local latitude and longitude on the navigation control panel in the MAN FIX position and then placing the position control to PRES POS. Magnetic heading is supplied by the flux valve and summed with manually inserted magnetic variation to derive true heading for platform coarse alignment.

With the platform control switch positioned to NORM, place the mode selector switch to HEAT. The HEAT lamp will illuminate, indicating system warmup. When the HEAT lamp is extinguished (2 to 3 minutes at ambient temperatures above 40°F), place the mode selector switch to ALIGN. This will illuminate the INEG STOP lamp and may cause the ERROR lamp to momentarily flicker.

CAUTION

> If the ERROR lamp remains illuminated, there is a malfunction in the INS. Turn the mode selector switch to OFF.

Approximately 90 seconds after going to ALIGN, the INEG STOP lamp will extinguish and the ALIGN lamp will illuminate. Steady illumination indicates coarse alignment and leveling have been completed and the platform is entering the gyrocompassing phase. The ALIGN lamp will start to flash within 5 to 10 minutes after entering ALIGN, indicating alignment is completed and ready for a navigation mode of operation.

If time permits, the degree of alignment may be improved by allowing the MAG HDG SYNC indicator to achieve as fine a null as possible prior to selecting a navigation mode. Before selecting a navigation mode, the position control selector must be repositioned from MAN FIX. In the MAN FIX mode, the position integrators are not engaged and the navigation computer will not maintain aircraft present position.

Note

> Taxiing or towing the aircraft while the platform is being aligned will delay the alignment and jeopardize its accuracy. After alignment is completed a NAV mode must be selected.

Rapid Alignment

Rapid alignment may be accomplished under shore-based conditions only. The decision to use this method is predicated on three conditions: urgency of accomplishing alignment rapidly, the accuracy of the gyrocompass heading previously stored in the system, and the aircraft remaining stationary from the time of the last alignment.

Use of this technique permits gyrocompassing prior to flight by ground alignment methods. After gyrocompassing has been completed and a fine null is achieved, true heading is stored in the navigation computer by placing the platform control to RAPID ALIGN. This mechanically locks the true heading shaft of the computer with the true north axes of the inertial platform. When power is returned to the system, this previously determined true heading is used as a reference and compared to the angular position of the platform azimuth. The difference is applied to the azimuth gyro motor to properly reposition the azimuth axis. Thus, the gyrocompassing phase is bypassed and rapid alignment can be accomplished in approximately 90 seconds.

During rapid align, the ALIGN lamp will remain out until alignment is completed at which time the lamp

will begin to flash. When rapid alignment has been completed, the platform alignment control should be repositioned to NORM before entering the GC or SR NAV mode.

CAUTION

If a platform failure occurs (ERROR lamp illuminated) and an AUX navigation mode is selected with the platform control in RAPID ALIGN, the true heading shaft in the computer will lock. This condition could cause damage to the servo amplifier in the heading module.

Carrier Alignment

Carrier alignment is accomplished similar to ground alignment; however, more time is required to level the platform because of the carrier's motion. When alignment is performed on the deck of a moving carrier true heading, latitude, longitude and velocity of the carrier are required inputs as initial conditions. The ship's inertial navigation system (SINS) provides this information to the aircraft platform through a carrier alignment receptacle in the left aft corner of the nose wheel well. This receptacle also provides a two-way communications link between aircraft ICS and SINS personnel. The carrier's relative velocity computer modifies velocity inputs to the INS as necessary, depending upon aircraft location with respect to the SINS. Aircraft spotting angle, which is the difference between aircraft true heading and carrier true heading, must be manually inserted in the relative velocity computer to provide corrected heading to the INS for course alignment.

When gyrocompassing is completed, the ALIGN lamp will flash, indicating the INS is ready for an operational navigation mode. Carrier alignment requires a minimum of twelve and one half minutes.

OPERATING NAVIGATION MODES

After the inertial platform has been aligned (indicated by a flashing ALIGN lamp and a null on the MAG HDG SYNC), the system is ready to enter an operating navigation mode. There are two system modes of navigation, normal mode using platform inputs or auxiliary mode used when the platform is inoperative Both modes consist of either a great circle (GC) or a short range (SR) mode of computing range and course to destination.

A great circle mode should be selected when the distance between present position and destination is greater than 200 nautical miles. In this mode the computer is programmed to solve equations based on a spherical triangle. Range and course are computed using the difference between inserted destination latitude and longitude and aircraft present position coordinates.

For missions when distance between present position and destination are within 200 nautical miles, a short range mode should be selected. In this mode the computer is programmed to solve equations based on a right triangle. The computer inputs (groundspeed, heading, airspeed, etc.) in GC and SR modes are exactly the same. Only the method of computing range and course angle differ.

An auxiliary navigation mode is automatically entered when platform signals are unreliable. When operating in this mode, flux valve magnetic heading is corrected by manually inserting magnetic variation to replace platform true heading. Airspeed is still supplied by the air data computer, but wind direction and velocity must be manually inserted. Equipped with these inputs, the computer provides aircraft navigation.

NAVIGATION CONTROL PANEL

The navigation control panel (figure 8-1) on the Missile Control Officer's (MCO) right console is a display and control unit. It provides the controls and indicators required to monitor and direct platform alignment, select navigation modes and perform computer self-test reliability ground checks. The panel displays latitude, longitude, speed, bearing, range, course and magnetic variation. The controls select functions to be performed by the navigation computer and provide slew controls for inserting position coordinates, wind components and magnetic variation.

Mode Selector Switch

The mode selector is a rotary switch on the navigation control panel. It provides turn-on power, warm-up and navigation operating modes. The switch has seven detents and requires a pullout to rotate from ALIGN to normal navigation modes (GC NAV or SR NAV), and from normal navigation modes to AUX navigation modes. Control positions and functions are as follows:

OFF - All electrical power is withheld from the inertial navigation system.

HEAT - Provides electrical power only to the platform heaters for gyro stabilization. Requires the platform control switch to be in RAPID ALIGN or NORM position. A HEAT lamp is provided to monitor the warm-up period.

ALIGN - Provides electrical power to the entire INS and initiates the platform alignment sequence when the platform control switch is in any position other than PLATFORM OFF. An align lamp is provided to monitor the progress of the platform alignment.

GC NAV - Long-range normal navigation mode used for ranges in excess of 200 nautical miles. The range and course computers solve for the great circle course and distance from present position to the destination.

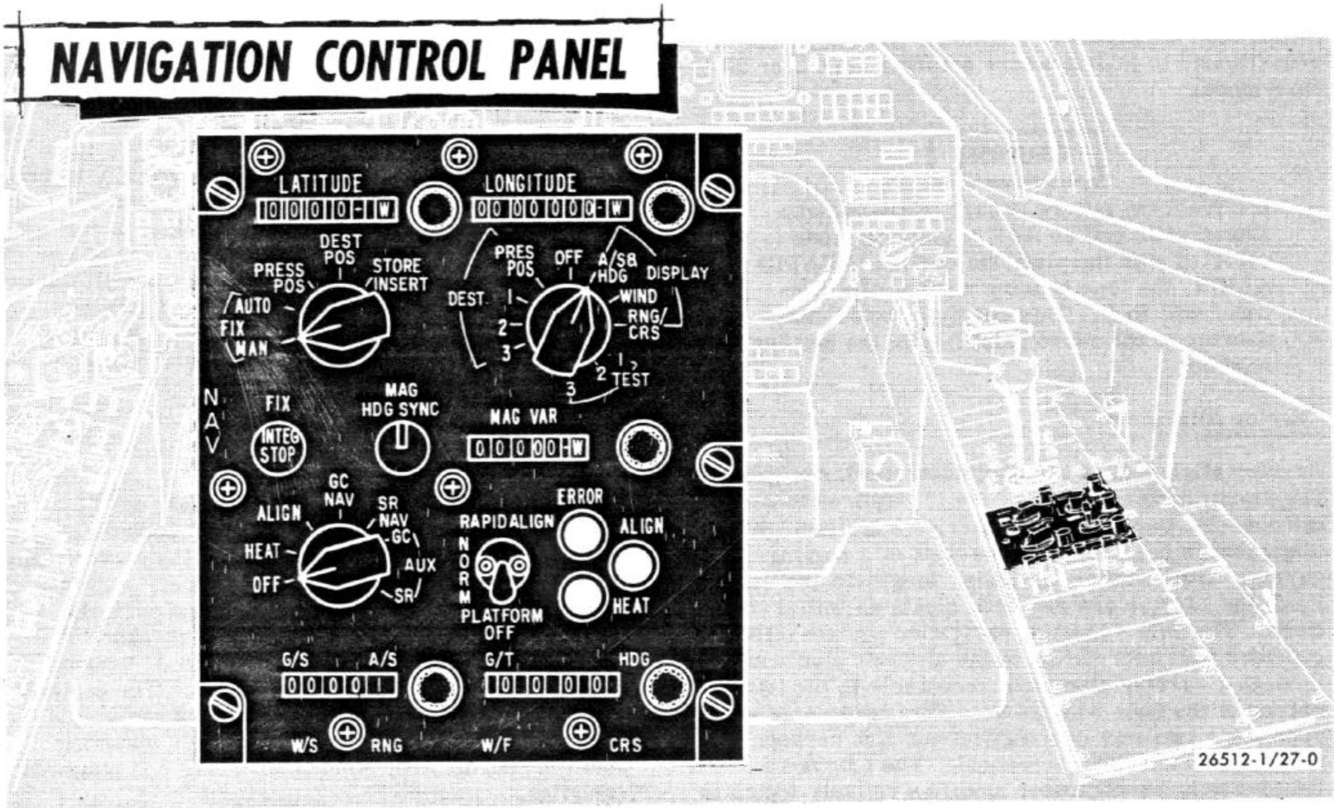


Figure 8-1.

SR NAV - Short-range normal navigation mode used within 200 nautical miles of destination, in which the range and course computers solve for a rhumb line course and distance from present position to destination.

AUX - Provides auxiliary navigation (GC and SR) modes automatically when inertial platform fails. This is indicated by illumination of the ERROR lamp. Positioning the mode selector switch to AUX mode will extinguish the ERROR lamp.

Platform Alignment Switch

The platform alignment switch is a three-positioned switch placarded PLATFORM OFF, NORM and RAPID ALIGN. A rotating guard is incorporated to ensure that the switch is locked in the selected position. Position of the switch provides alignment mode selection (normal or rapid) of the inertial reference unit.

The OFF position deenergizes the inertial platform. The NORM position is used for ground alignment, carrier alignment, and operating in any normal or auxiliary navigation mode. RAPID ALIGN position is used to preset aircraft heading in the navigation computer so that when the system is reactivated the platform will align to this stored heading, providing the aircraft has not been moved.

Platform Indicator Lamps

Three platform indicator press to test lamps on the navigation control panel (figure 8-1) are placarded HEAT, ALIGN, and ERROR. These are advisory indicators and provide the MCO with a visual means of monitoring platform operation.

HEAT LAMP - An amber lamp provides a visual monitor of the platform warm-up temperature when the system is energized. The lamp will illuminate until the platform reaches minimum ambient temperature for gyro stabilization (above 15° F). The platform will not enter the gyrocompass phase until the lamp is extinguished.

ALIGN LAMP - A green lamp permits the MCO to monitor the progress of the platform alignment. With the platform control switch in NORM, a steady illumination indicates coarse alignment, and leveling is completed. The platform is now in the gyrocompass phase. Approximately 55 to 90 seconds will

elapse after selecting the ALIGN mode and the lamp illuminates. A flashing lamp indicates gyro-compassing is completed and the system is ready for the operating NAV mode.

With the platform control switch in RAPID ALIGN, the align lamp will remain out until alignment is complete, at which time the lamp will begin to flash.

ERROR LAMP - In flight this amber lamp indicates the inertial reference unit has malfunctioned when the mode selector is in ALIGN or either of the normal NAV positions. The ERROR lamp will remain illuminated until an AUX navigation mode is selected. A discrete indicator lamp on the VDIG will also illuminate with PLATFORM in black letters.

Magnetic Heading Synchronization Indicator

The magnetic heading synchronization indicator is placarded MAG HDG SYNC. In the normal navigation modes, it reflects the error between computed magnetic heading (platform true heading and inserted magnetic variation) and magnetic heading from the flux valve, indicating that the correct magnetic variation is not inserted. During normal navigation modes, the meter should be maintained at null by periodic manual correction of the magnetic variation to correct computed magnetic heading used by the flight instruments.

In the AUX navigation modes, only flux valve headings are used by the computer and flight instruments, and the MAG HDG SYNC should be at a null (±1/2 indicator bar width). In this mode the indicator reflects the degree of agreement between the INS flux valve and AFRS magnetic heading.

In the platform alignment modes, the indicator is a also used to monitor the progress of the gyrocompassing phase. Allowing the meter to achieve as fine a null as possible before switching to a navigation mode will improve the degree of alignment. The quality of alignment is proportional to the quality of the null.

Magnetic Variation Indicator (MAG VAR)

The magnetic variation indicator placarded MAG VAR (figure 8-1) on the navigation control panel is used to display manually inserted (East or West) magnetic variation. The magnetic variation is manually inserted by a manual control adjacent to the indicator.

Position Control Selector

The position control selector is a five-positioned, rotary control switch. It selects readouts for display on the latitude and longitude position counters (present position or destination), controls updating present position, destination and stored coordinates, and controls the overfly updating modes. The switch has detents at all positions and requires a pullout to rotate from AUTO FIX to PRES POS. The control positions and functions are as follows:

MAN FIX - Decouples position integrators so that they may be manually advanced to coordinates of an upcoming fix point. The lamp on the FIX pushbutton switch illuminates INTEG STOP, indicating the position integrators of the navigation computer are inoperative. In ground alignment procedures this position is used to initially insert aircraft present position coordinates using the latitude and longitude slew controls.

AUTO FIX - Provides automatic updating present position with inserted destination coordinates by depressing the FIX pushbutton when over the fix point. In this mode the position integrators do not stop and coordinates continue to integrate aircraft velocity.

PRES POS - Provides for display of aircraft present position coordinates from the navigation computer in the latitude and longitude readout windows. Slew controls are disabled in this position.

DEST POS - Provides display of the destination coordinates from the navigation computer in the latitude and longitude readout windows. Slew control knobs are provided to manually set destination coordinates in the navigation computer.

STORE INSERT - Enables insertion of latitude and longitude coordinates into any of the three destination storage channels of the navigation computer.

Latitude and Longitude Indicators

The two indicators placarded LATITUDE and LONGITUDE display present position or destination coordinates in degrees and minutes. The position control selector determines the readout (present position or destination) displayed. During normal modes of navigation, the present position coordinates are continuously and automatically updated by inputs of true north and east velocity components from the inertial platform.

During AUX navigation modes, coordinates are similarly updated by north and east velocities derived from airspeed, wind data, handset or stored heading determined from the flux valve, and manually inserted magnetic variation.

When the position control selector is in STORE INSERT, the indicators display one of the three

destinations stored and selected by the DEST/DISPLAY/TEST selector.

Slew controls with fast or slow speed control are provided for handsetting the counters with the initial coordinates or to insert manual correction.

Integrator Stop Lamp and Fix Pushbutton

The combined integrator stop lamp and fix pushbutton placarded FIX is used in conjunction with overfly fix modes of the position control selector. In MAN FIX mode, the lamp illuminates and the words INTEG STOP are visible, indicating the computer integrators have stopped. Depressing the button reenergizes the integrators and extinguishes the lamp. In AUTO FIX mode, the integrators are not stopped and aircraft velocity continues to be integrated by the computer while updating. When over the fix point, depressing the button updates present position by driving the present position to the inserted destination coordinates. Since the coordinates of the fix point have been previously inserted into the destination position, present position is corrected.

DEST/DISPLAY/TEST Selector (DDT Switch)

The destination display test selector is a rotary control switch that provides selection of stored destination coordinates for display on the latitude and longitude indicators. It also provides selection of three self-test modes to ground check computer and control panel reliability. The control positions and functions are as follows:

DEST 1, 2, 3 - Operates in conjunction with the STORE INSERT on the position control selector. Selects the destination coordinates for insertion and storage in the computer and displays them in the latitude and longitude indicators. Groundspeed (G/S) and ground track (G/T) are displayed in the speed and bearing indicators.

PRES POS - Used with the STORE INSERT on the position control selector to change destination coordinates to present position. Groundspeed (G/S) and ground track (G/T) are displayed in the speed and bearing indicators.

OFF - Normal operating position of the switch. Groundspeed (G/S) and ground track (G/T) are displayed on the speed and bearing indicators.

DISPLAY A/S-HDG - Selects true airspeed (A/S) and true heading (HDG) from the computer for display on speed and bearing indicators.

DISPLAY WIND - Selects computed wind direction (W/F) and velocity (W/S) for display in the speed and bearing indicators. In the AUX navigation modes, wind speed (W/S) and direction (W/F) may be slewed and inserted in the navigation computer.

DISPLAY RNG/CRS - Selects the computed range (RNG) and true course (CRS) to destination for display in the speed and bearing indicators.

TEST 1, 2, 3 - Used for ground tests to check navigation computer reliability and associated readouts.

CAUTION

Do not select TEST modes in flight.

Speed and Bearing Indicators

The individual speed and bearing indicators on the lower section of the navigation control panel are not placarded. The position of the DDT selector determines the readout displayed. All readouts represent data in the navigation computer. When OFF (normal operation), PRES POS or DEST 1, 2, 3 are selected, groundspeed (G/S) and ground track (G/T) are displayed. In DISPLAY A/S-HDG, the true airspeed (A/S) and true heading (HDG) are displayed. Selecting DISPLAY WIND displays wind direction (W/F) and wind velocity (W/S). In the AUX navigation modes, wind speed and direction may be slewed into the computer. With DISPLAY RNG/CRS selected, the range (RNG) to destination and true course (CRS) are displayed. Display indicator lamps above and below the counters illuminate with the abbreviated symbol verifying the particular readout displayed. Slew controls are provided for handsetting the counters with wind data in the auxiliary navigation modes.

PRESENT POSITION UPDATING

The operation of any navigation system over a long period of time will result in errors in the system. The inertial navigation system is no exception. Therefore, depending upon the mission, updating present position coordinates for accurate navigation may be required.

Manual Updating

This method of updating requires flying the aircraft over a check point. Approaching the fix point, place the position control selector to MAN FIX. This stops the position integrators in the computer and is indicated by illumination of INTEG STOP on the FIX pushbutton lamp. Using the latitude and longitude slew controls, change the present position coordinates to those of the upcoming check point. When the aircraft is over the check point, depress the FIX pushbutton. This restarts the position integrators and extinguishes the lamp. Return the position control selector to PRES POS and check the coordinates.

Automatic Updating

Automatic updating is performed similar to manual updating. Both methods require flying the aircraft

over a check point and depress the FIX pushbutton. As the aircraft approaches the fix point, insert the coordinates of this point into the destination channel of the computer using the latitude and longitude slew controls if necessary. If the coordinates are in one of the computer's storage positions, they can be inserted directly by selecting DEST 1, 2, or 3 on the DDT selector. When the coordinates have been inserted, place the position control selector to AUTO FIX. In this mode, the position integrators do not stop. When over the check point, depress the FIX pushbutton. This corrects the present position counters to the same coordinates as the destination position selected, and the integrators continue to integrate aircraft velocity. The integrators are updated by an amount equal to the difference between the destination and present position at the time the pushbutton is depressed. This updating can be observed on the latitude and longitude windows. When updating is completed, return the position control selector to PRES POS.

VERTICAL DISPLAY INDICATOR GROUP (VDIG) AN/AVA-3

The vertical display indicator group system (figure 8-2) is the pilot's primary integrated flight control indicator, providing an all-weather attack capability independent of visual flight conditions. Centered on the instrument panel directly in front of the pilot, it provides a television type display for optimum control during takeoff, cruise, terrain avoidance, attack and landing. Information to present this integrated comprehensive display of flight data is received from the peripheral sensor equipments including AMCS, CADC, INS, DATA LINK, TID, and radar altimeter. These inputs are converted to electronically-generated image symobols used in various combinations for for each mode of operation, viewed on two displays (figure 8-3): a direct view indicator (DVI) and a projection indicator (PI). Although the DVI and PI are physically connected, they are completely electrically isolated so that a failure in either will not affect the other. In addition to two indicators, the group includes a turn and slip indicator, a radar altitude indicator, a true-airspeed indicator, and four discrete indicator lamps (WAVE OFF, MISSED MESSAGE, RADAR ALT LOW, and PLATFORM).

DIRECT VIEW INDICATOR (DVI)

The direct view indicator (DVI) is a TV raster-type presentation that simulates the view seen through the aircraft windshield with angular visibilities of 60 degrees elevation and 75 degrees azimuth. Actual flight contact is represented in analog form by a field of ground texture and sky texture separated by a horizon line. It provides primary aircraft attitude and selected command flight information and is capable of displaying, simultaneously or separately, a tactical situation and weapons delivery data. The DVI displays a continuous true presentation of the change in attitude through 360 degrees of roll and azimuth and +90 to -90 degrees of pitch. Two manual controls on the sides and below the indicator permit adjustment of brightness and pitch trim. Attached to the indicator face is a micromesh filter which provides improved daytime viewing and serves as a protective mask against implosion of the CRT. An aviation-red night filter is available for retention of the pilot's night vision adaptation. Controls for selecting operational modes are on the display control panel (figure 8-4).

PROJECTION INDICATOR (PI)

The projection indicator consists of a transparent combining glass with visual symbols optically projected and superimposed upon the real world seen through the aircraft windshield. This is a specialized display providing basic flight information for precise control of the aircraft without reference to the instrument panel. Symbols are displayed through optics collimated at infinity and provide minimum interference with the pilot's outside vision. The PI has a field of view of approximately 12 degrees in elevation and 10 degrees in azimuth. Within this view, the pilot has a continuous presentation of the change in attitude through 360 degrees of roll and +10 to -10 degrees of pitch. The brightness of the symbols displayed is automatically controlled in accordance with the level of the ambient light around the indicator. Controls for selecting operational modes are located on the display control panel.

DISCRETE INDICATORS

Four discrete indicator lamps are grouped on the center section of the VDIG (figure 8-3). When energized, the lamps illuminate red with the associated discrete message in black letters. Lamp circuitry may be tested by depressing the TEST button on the master lighting panel.

WAVE OFF - A flashing light indicates wave off in ACL mode and energizes the breakaway symbols on the DVI and PI displays.

MISSED MESSAGE - Indicates no message received in 10 seconds from DATA LINK. In the ACL mode, illumination indicates no message received in 2 seconds and energizes the breakaway symbols on the DVI and PI displays.

RADAR ALT LOW - Indicates radar altitude above the terrain is less than that set on the radar altimeter limit index.

PLATFORM - Indicates the stabilization platform of the inertial navigation system is inoperative or malfunctioning.

DISPLAY CONTROL PANEL

The display control panel (figure 8-4) is on the pilot's left console. It provides controls for selecting flight, command and tactical analog presentations on the VDIG displays and the tactical information display (TID). At the present time only those control functions associated with flight mode presentations on the VDIG are operational.

PILOT'S VERTICAL DISPLAY SYSTEM

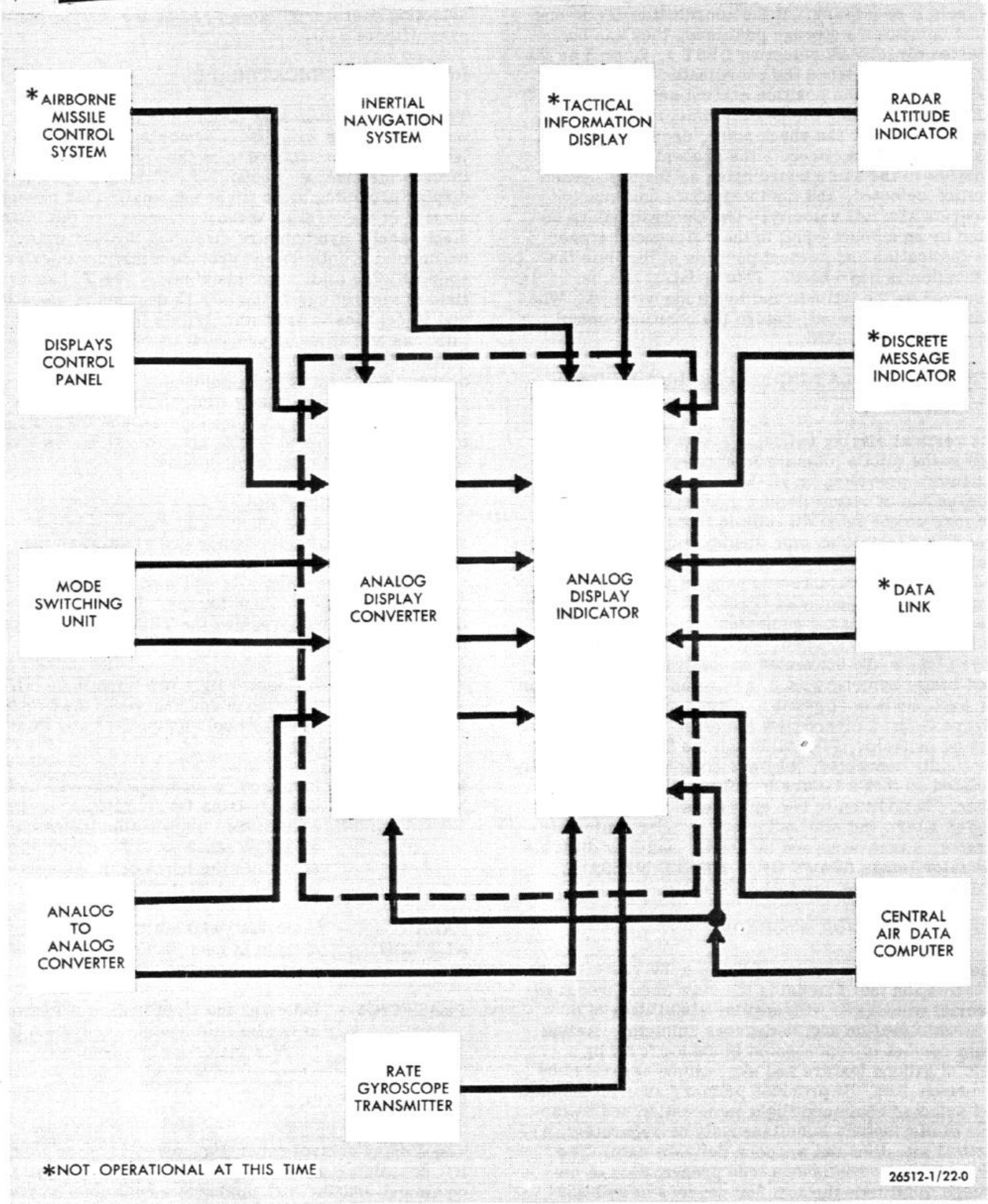


Figure 8-2.

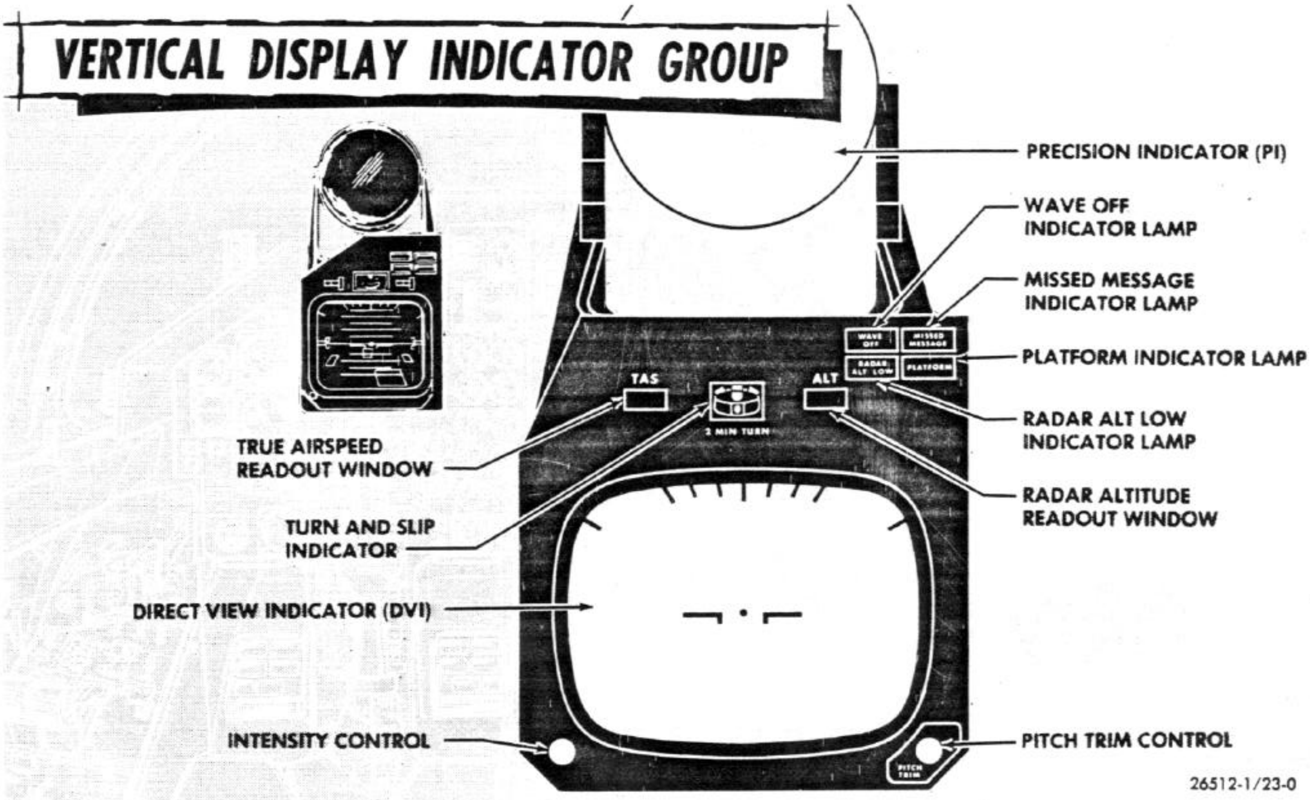


Figure 8-3.

V/HUD MODE Selector Switch

The V/HUD MODE selector is an 11-positioned rotary switch that provides operating mode selection for presentation on the DVI and the PI displays only. Control positions and functions are as follows:

OFF - All electrical power is withheld from the DVI and PI, TAS indicator and radar altitude indicator on the VDIG. Power for the four discrete indicacators is not affected.

ALT WPN - The alternate weapons mode has no operational function.

ATT ONLY - Selecting the attitude only mode provides roll, pitch and trim information only. This mode should be selected when steering data (INS unreliable) is unavailable.

T.O. - In the takeoff mode, the angle of attack error symbol is not abailable on either display. On the PI, the radar altitude scale and true airspeed scale are available.

FLT - The flight mode is the basic mode of operation. All symbols are available on the DVI except angle-of-attack. On the PI, angle-of-attack, true airspeed scale and radar altitude scale are not available.

LDG - In the landing mode, the angle-of-attack error symbols are available on both displays. On the PI display, radar altitude and true airspeed scales are also presented.

CAUTION

Selecting TA, TAC/I, or TAC STBY for more than 15 seconds will damage the equipment.

TA - Terrain avoidance mode has no operational function.

TAC/I - Tactical mode 1 has no operational function.

TAC STBY - Tactical standby mode has no operational function.

TEST 1, 2 - Test modes 1 and 2 are display pattern arrangements to determine if a particular symbol is available within the circuitry and generally positioned properly. Symbols do not always appear in an exact position; however, the DVI pattern test mode displays symbols in groups of three. Thus, the pilot can become familiar with the test pattern symbols. Test mode 2 presents the DVI display

DISPLAYS CONTROL PANEL

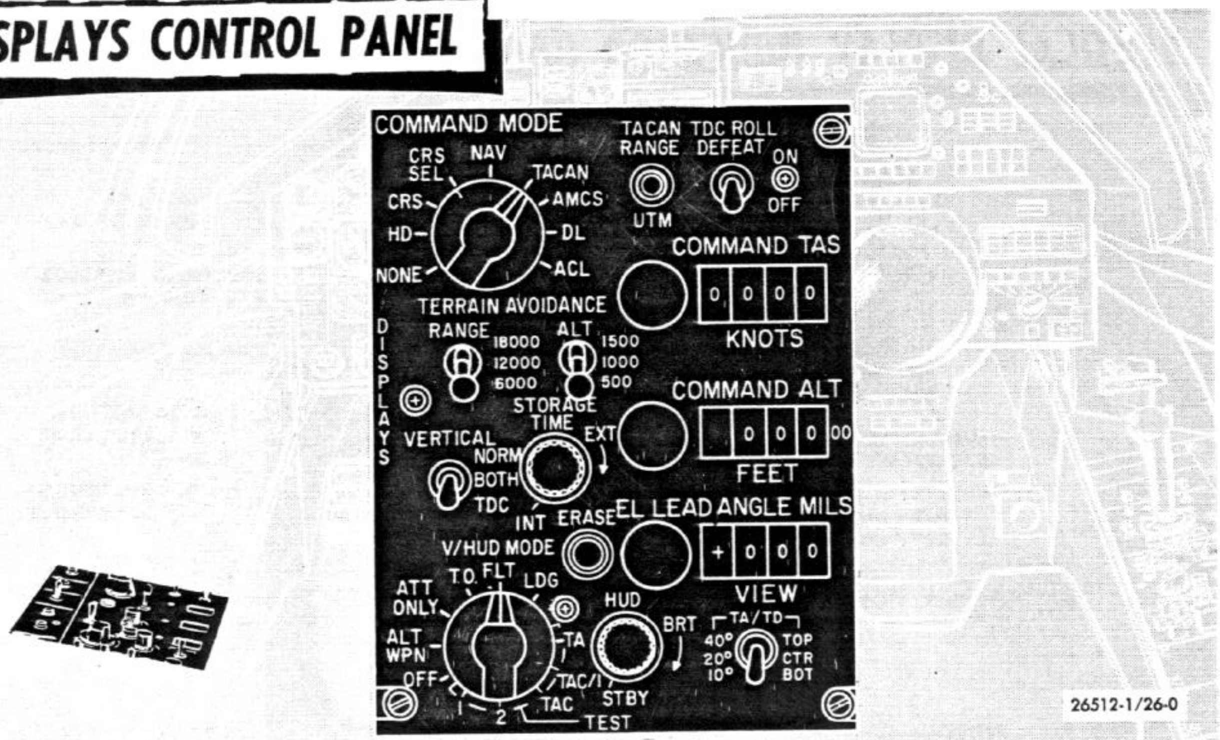


Fig. 8-4

rolled 30 degrees. The test pattern symbols are valid only in test mode 1 on the PI.

Note

Regardless of the position of the COMMAND MODE switch, selecting TEST 1 or TEST 2 position on the V/HUD MODE switch will display a test pattern on the VDIG indicators.

VERTICAL Switch

The VERTICAL switch is a three-positioned toggle switch placarded NORM, BOTH, and TDC. At the present time, only the NORM position is operational and should be selected to obtain VDIG displays.

COMMAND MODE Selector Switch

The COMMAND MODE selector is a 9-positioned rotary switch. It allows the pilot to select the subsystem from which input data is desired for display in various combinations of operating modes. The present aircraft configuration provides flight data information on the VDIG and HSI only. Control positions and functions are as follows:

NONE - In this position no steering symbols are available on either VDIG indicator. On the HSI, the course deviation bar is caged in the center of the indicator. This position should be selected if the INS is unreliable.

HD - The heading mode allows the pilot to fly a manual selected magnetic heading. This option is available with the INS operational or inactive. When the INS is inoperative, pitch, roll, and magnetic heading signals are supplied through the auxiliary flight reference system (AFRS) from the standby compass. Heading is selected by turning the HDG SET knob on the HSI and aligning the heading marker opposite the selected heading on the compass card. Steering error symbols on the VDIG displays and the deviation bar on the HSI indicate the same heading error. This heading error is the angular difference between manual selected heading and aircraft magnetic heading. If TACAN is on and operating, the No. 2 pointer and MILES counter on the HSI display TACAN information.

CRS or CRS SEL - The course modes provide a capability to fly a manually selected course, although the INS continues to compute ground track. A desired course is selected by turning the CRS SET knob on the HSI until the course selected is displayed on the digital course readout window. Steering error symbols on the VDIG displays and the HSI course deviation bar represent the difference between the INS ground track and the manually selected course. If TACAN

is on and operating, the No. 2 point and MILES counter on the HSI display TACAN information.

NAV - The navigation mode is the basic operating mode. The INS provides heading, computed course and actual ground track to the HSI. The course deviation bar and steering error symbols indicate heading error caused by changes in wind velocity and direction. If the INS platform malfunctions, magnetic heading is automatically received from the standby compass system. On the HSI, the No. 2 pointer will indicate TACAN bearing while the MILES counter will display distance to destination, computed by the INS. TACAN fixing is available by use of the TACAN RANGE/UTM switch.

TACAN - This mode provides flying a selected course referenced to a TACAN radio facility. Steering error and course deviation bar deflection represent difference between INS computed course and TACAN steering information.

AMCS - The airborne missile control system mode has no operational function

DL - The data link mode has no operational function.

ACL - The automatic carrier landing mode has no operational function.

TACAN RANGE/UTM Switch

The TACAN RANGE/UTM switch is a two-positioned, momentary, toggle switch. The TACAN RANGE position is used with the command mode switch in the NAV or CRS SEL position. While the switch is engaged, TACAN range can conveniently be read on the range counter of the HSI. The UTM position, associated with data link, is not operational.

Note

During normal flight operation (INS operational and NAV mode or CRS SEL selected), the range counter on the HSI will display distance from present position to a predetermined destination position computed by the navigation computer.

COMMAND TAS Indicator and Control

Airspeed for the command airspeed error symbols are selected by pushing the COMMAND TAS control knob to the inner detent and turning. The selected airspeed values can be read on the adjacent readout window. Pulling the control knob to the out detent disconnects the circuit supplying TAS from the CADC to the VDIG.

COMMAND ALT Indicator and Control

Altitude for the command airspeed error symbols are selected by pushing the ALT COMMAND control knob to the inner detent and turning. The selected altitude can be read on the adjacent readout window. Pulling the control knob to the outer detent disconnects altitude signals to the VDIG.

Manual Brightness Control (HUD BRT)

The HUD BRT control knob provides manual adjustment for PI symbols and must be in a clockwise position to establish a nominal value for operation of the automatic brightness control circuit.

TDC ROLL DEFEAT Switch

This switch has no operational function.

TERRAIN AVOIDANCE Switches

The RANGE and ALT terrain avoidance switches have no operational function.

STORAGE TIME Control

This switch has no operational function.

ERASE Button

The erase button has no operational function.

ELEVATION LEAD ANGLE MILS Indicator

This indicator has no operational function.

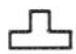
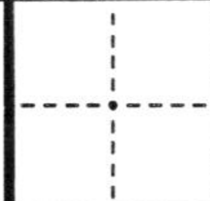
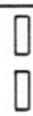
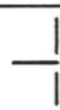
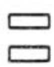
VIEW TA/TD Switch

This switch has no operational function.

SYMBOLOGY

Symbology presented on the VDIG displays is determined by the operating mode selected on the display control panel (figure 8-4). The three switches on the display control panel whose settings determine the symbols presented on the displays are: COMMAND MODE switch, V/HUD MODE switch, and VERTICAL switch. The displays, developed and generated by an analog display converter for presentation on the DVI and PI, are electrically separate and independent. The various symbols used with each display, redundant to some extent in name and meaning, differ in shape and presentation. Figure 8-5 lists the name of each symbol and illustrates its DVI and PI configuration. The shade of a particular symbol is fixed; however, the position of the symbol on the display is variable. DVI symbol shades are black, dark gray, medium gray, light gray or white (maximum brightness). Figure 8-6 (sheet 1 through 6) combinations of symbols that appear on the DVI and PI displays.

VDIG DISPLAY SYMBOLS

NOMENCLATURE	DVI	PI
STEERING SYMBOL		
MOVEABLE RETICLE	NONE	
AIRCRAFT RETICLE (PAINTED DVI) (ELECTRONIC PI)		
COMMAND SPEED ERROR		
COMMAND ALTITUDE ERROR		
TIME AND RANGE		
BREAKAWAY		
FIXED RETICLE	NONE	
GROUND TEXTURE ELEMENTS		NONE

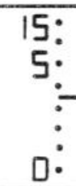
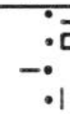
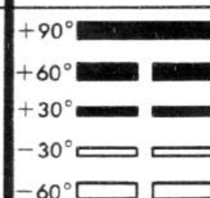
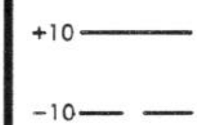
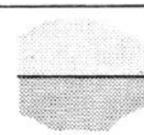
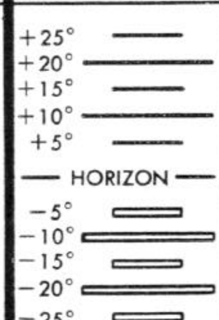
NOMENCLATURE	DVI	PI
ANGLE OF ATTACK		
RADAR ALTITUDE SCALE	NONE	
TRUE AIRSPEED (SCALE)	NONE	
PRECISION COURSE VECTOR (PCVS)		NONE
IMPACT POINT		
TARGET SYMBOL		NONE
MAJOR PITCH LINES		
HORIZON LINE		
INCREMENTAL PITCH LINES + (BLACK) − (WHITE)		NONE

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Figure 8-5.

Certain symbols have the potential of appearing at all times in all modes, but require discrete signals to activate them. Others, such as impact point, precision course vector and major pitch lines, appear in test modes, but are not included in the operating mode illustrations. This indicates they are potentially available within the circuitry but are not associated with a particular mode. Major pitch lines are shown only for applicable pitch attitudes. Since the illustrations (other than Test Mode 2) show zero pitch, the lines are off the display.

Aircraft Reticle

The aircraft reticle consists of two black lines representing an aircraft with gear down and a dot in the center. On the DVI, the reticle is permanently inscribed on the indicator face and when the horizon is aligned with the "wings" of the reticle, the aircraft is in level flight. The reticle on the PI is shaped identical to the DVI symbol, but is electronically generated and fixed at the center of the indicator.

Angle-of-Attack Error

On the DVI, the angle-of-attack error symbol is a bright hexagon. Used in the landing modes, it moves vertically with reference to the port "wing" of the aircraft reticle. If the symbol is below the aircraft reticle, angle-of-attack is too high; above the aircraft reticle indicates angle-of-attack is too low. The angle-of-attack error symbol for the PI consists of a short vertical bar and is indicated in the same manner as on the DVI.

Note

A blinking symbol is warning that a stall condition is imminent.

Ground Texture

The ground texture symbol for the DVI consists of a dark-gray field with black trapezoids which differentiate it from the uniform light-gray sky texture. The size and spacing of the trapezoids are arranged to give perspective to the presentation. Ground texture remains parallel to actual horizon and provides a basic aircraft attitude reference compatible with heading change. When the aircraft is moving and ground texture is visible, the trapezoids emanate from the horizon and move at a fixed rate to simulate motion. There are no ground texture symbols necessary for the PI since the real world serves this purpose.

Horizon Line

This is a demarcation point between ground and sky textures presented by a sudden brightening on the display where ground texture ceases. It represents the real horizon, is ground stabilized and changes orientation with any change in aircraft pitch or roll. On the PI, this symbol is presented as a black line and is also stabilized to reflect aircraft pitch and roll.

Command Airspeed Error

The DVI command airspeed error symbol at the extreme left of the display consists of a fixed reference mark and two vertically movable reference bars. Bars below the fixed reference mark indicate the aircraft is below command airspeed. Bars above the fixed reference mark indicate the aircraft is above command airspeed. On the PI, this symbol is identical except that the elements are thinner.

Command Altitude Error

The DVI command altitude error symbol is similar to the command airspeed error symbol except it is located at the extreme right of the display and the two vertically movable bars are wider horizontally. Bars below the reference mark indicate the aircraft is above the command altitude. Bars above the fixed reference marker indicate the aircraft is below the command altitude. The PI command altitude symbol is identical to the DVI except that the elements are thinner.

Impact Point

Impact point on the DVI is represented by a bright circle with a dark outline. It indicates a point the aircraft will intercept if it continues on its present course. The PI impact point appears similar to the DVI symbol except that it has no outline.

Major Pitch Lines

The major pitch lines for the DVI comprise +90, +60, +30, 60, and 30 degree pitch lines. The 30 and 60 degree pitch lines are dashed and appear black above the horizon and bright below. Since the elevation field of view for the DVI is 60 degrees, only two pitch lines (except in a test mode) can appear at one time. When a particular pitch line is at the vertical center of the display, it indicates the pitch angle of the aircraft. The PI has two major pitch lines: +10 and -10 degrees. Since the elevation field of view for the PI is 12 degrees, only one pitch line (except in a test mode) can appear at one time.

Intermediate Pitch Lines

Intermediate pitch lines on the DVI appear at 5 degree intervals for 25 degrees above and 25 degrees below the horizon line (0-degree pitch line). There are no intermediate pitch lines for the PI display.

Radar Altitude

There is no radar altitude symbol for the DVI. However, the radar altitude indicator is adjacent to the VDIG for reference. On the PI, the radar altitude symbol consists of a movable pointer and a fixed altitude scale on the extreme right of the glass. This is a nonlinear scale divided into 100-feet increments from 0 to 500 feet, and in 500 feet increments from 500 to 1500 feet. The radar altitude symbol will appear only in the TAKEOFF and LANDING modes. It is turned off when the radar altimeter is unreliable or the radar altitude is over 5000 feet.

Steering

Both the DVI and the PI steering symbol are identical. It consists of a bright inverted T. Aircraft steering is accomplished by aligning and maintaining the vertical and horizontal bar of the inverted T with the aircraft reticle center dot.

True Airspeed

There is no true airspeed symbol for the DVI. On the PI display, a true airspeed scale consists of a movable pointer and fixed scale on the extreme left of the glass marked in 50-knot increments. This scale appears only in the landing and takeoff modes.

Time/Range

On the DVI, time/range is presented by a bright, hollow, diamond-shaped symbol with a black segment as a second hand. This black segment moves counterclockwise from the 11 o'clock position, indicating the time remaining until in-range. For delivery of secondary armament, the symbol represents range and the black segment movement indicates successive stages in the delivery sequence. The PI symbol is similar except it is octagon-shaped and has a gap in place of the black segment.

Fixed and Movable Reticles

Only the PI display contains fixed and movable reticles. These symbols serve as an optical sight for delivering weapons against ground targets. A set of cross hairs (fixed reticle) represents the armament datum line (ADL) of the aircraft. The movable reticle (dashed cross hairs) can be adjusted for a desired elevation lead angle.

Breakaway

The DVI and PI breakaway symbol is a large X that can blink at a rate of 2 cycles per second. It indicates wave off, missed message and pull-up or minimum range.

Target

The target symbol appears only on the DVI and consists of a bright square with a dark outline. It is used with certain weapons deliveries.

Precision Course Vector

The symbol consists of a dark vertical and horizontal bar. It appears only on the DVI and is used during carrier landings. When the bars form a cross at the center of the display, the aircraft is on course. Corrections are made in the direction of the respective bar.

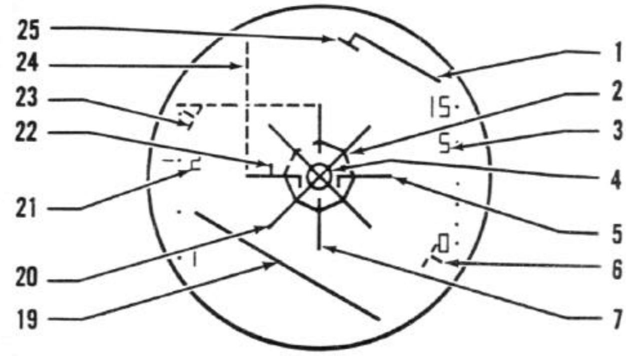
Roll Pointer

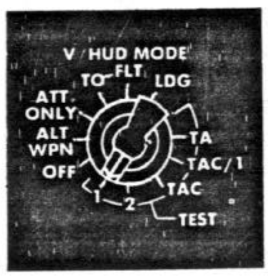
Only the DVI display contains this symbol consisting of a bright line extending down 1 inch from the top of the indicator at zero degree roll. Nine black roll markers are permanently marked on the indicator, four on each side of the zero degree marker positioned at top center. The first three markers are positioned at 10, 20 and 30 degrees and the fourth at 60 degrees.

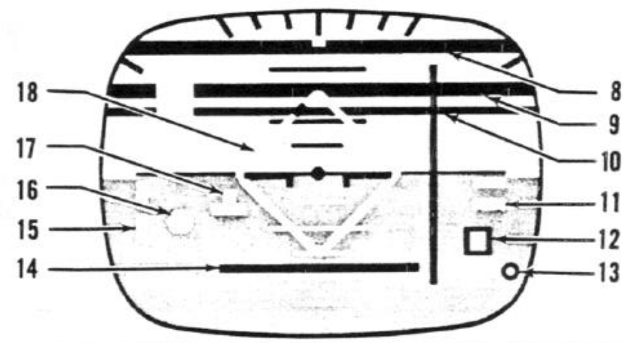
VDIG OPERATING MODES

TEST MODE 1

1. +10 DEGREE PITCH LINE
2. TIME/RANGE
3. RADAR ALTITUDE
4. IMPACT POINT
5. AIRCRAFT RETICLE
6. COMMAND-ALTITUDE ERROR
7. FIXED RETICLE
8. 90 DEGREE PITCH LINE
9. 60 DEGREE PITCH LINE
10. 30 DEGREE PITCH LINE
11. COMMAND-ALTITUDE ERROR
12. TARGET
13. IMPACT POINT
14. INTERMEDIATE PITCH LINE (—20 DEGREE)
15. COMMAND-AIRSPEED ERROR
16. ANGLE-OF-ATTACK ERROR
17. STEERING
18. TIME/RANGE
19. HORIZON LINE
20. BREAKAWAY
21. TRUE-AIRSPEED SCALE
22. ANGLE OF ATTACK ERROR
23. COMMAND-AIRSPEED ERROR
24. MOVEABLE RETICLE
25. STEERING





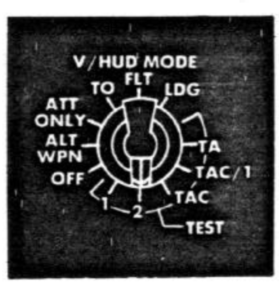


TEST MODE 2

1. ROLL INDICES (0°, 10°, 20°, 30°, 60°)
2. ROLL POINTER
3. INTERMEDIATE PITCH LINE (+5 DEGREE)
4. SKY TEXTURE
5. HORIZON
6. GROUND BACKGROUND
7. GROUND-TEXTURE ELEMENT
8. REFERENCE MARKER
9. —60-DEGREE PITCH LINE
10. BREAKAWAY
11. —30-DEGREE PITCH LINE
12. AIRCRAFT RETICLE

NOTE

DISREGARD ALL PI SYMBOLS IN TEST MODE 2.



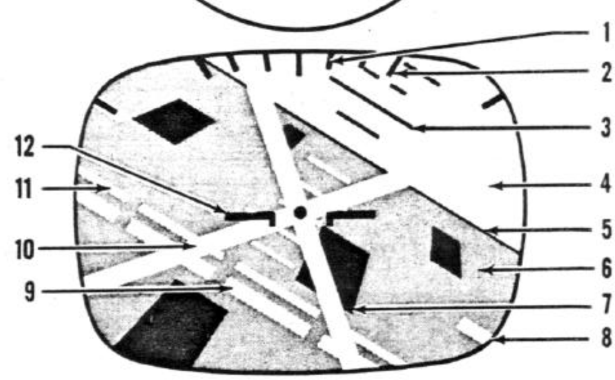


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Figure 8-6. (Sheet 1)

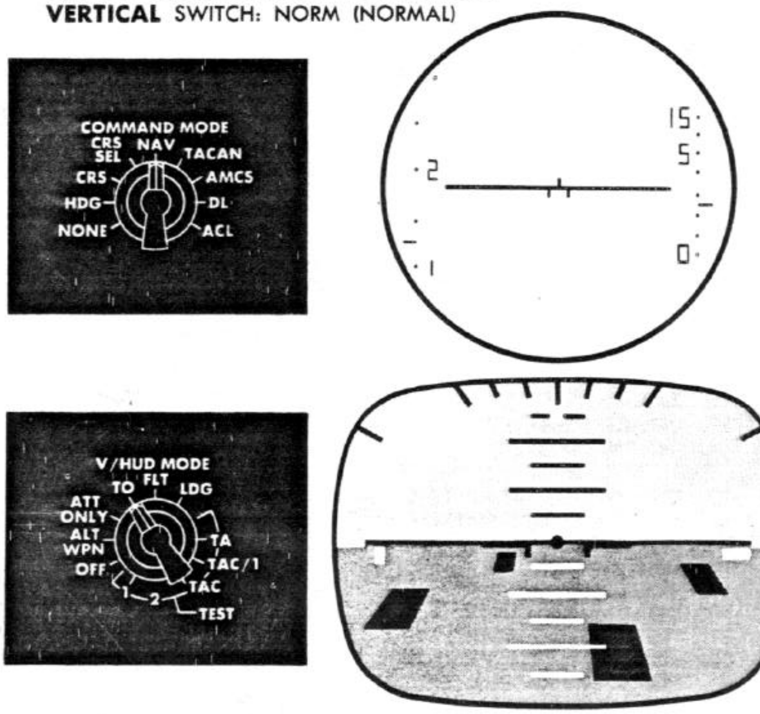
VDIG OPERATING MODES Continued

COMMAND MODE SWITCH: ANY NAVIGATION MODE **
V/HUD MODE SWITCH: T.O. (TAKEOFF)
VERTICAL SWITCH: NORM (NORMAL)

SYMBOL	DISPLAY DVI	DISPLAY PI
STEERING	X	X
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X*	
COMMAND ALTITUDE ERROR	X*	
TIME/RANGE		
BREAKAWAY		
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X	
ANGLE OF ATTACK		
RADAR ALTITUDE		X
TRUE AIRSPEED		X
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET		
INTERMEDIATE PITCH LINES	X	

NOTE

X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY OTHER AIRCRAFT SUBSYSTEM EQUIPMENTS.
** = HDG (HEADING), CRS (COURSE), CRS SEL (COURSE SELECT), NAV (NAVIGATION) OR TACAN



COMMAND MODE SWITCH: ANY NAVIGATION MODE **
V/HUD MODE SWITCH: FLT (FLIGHT)
VERTICAL SWITCH: NORM (NORMAL)

SYMBOL	DISPLAY DVI	DISPLAY PI
STEERING	X	X
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X*	X*
COMMAND ALTITUDE ERROR	X*	X*
TIME/RANGE		
BREAKAWAY		
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X	
ANGLE OF ATTACK		
RADAR ALTITUDE		
TRUE AIRSPEED		
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET		
INTERMEDIATE PITCH LINES	X	

NOTE

X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY OTHER AIRCRAFT SUBSYSTEM EQUIPMENTS.
** = HDG (HEADING), CRS (COURSE), CRS SEL (COURSE SELECT), NAV (NAVIGATION) OR TACAN

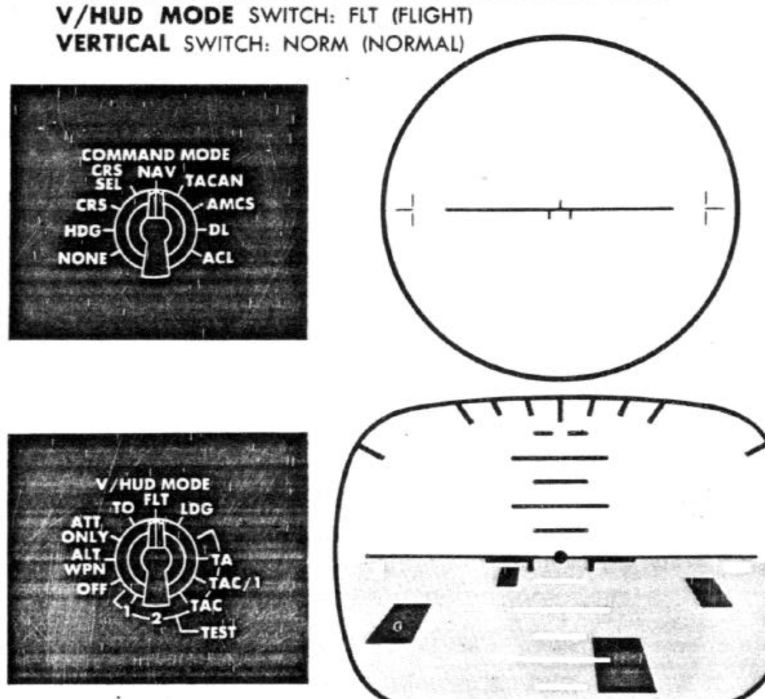


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Figure 8-6. (Sheet 2)

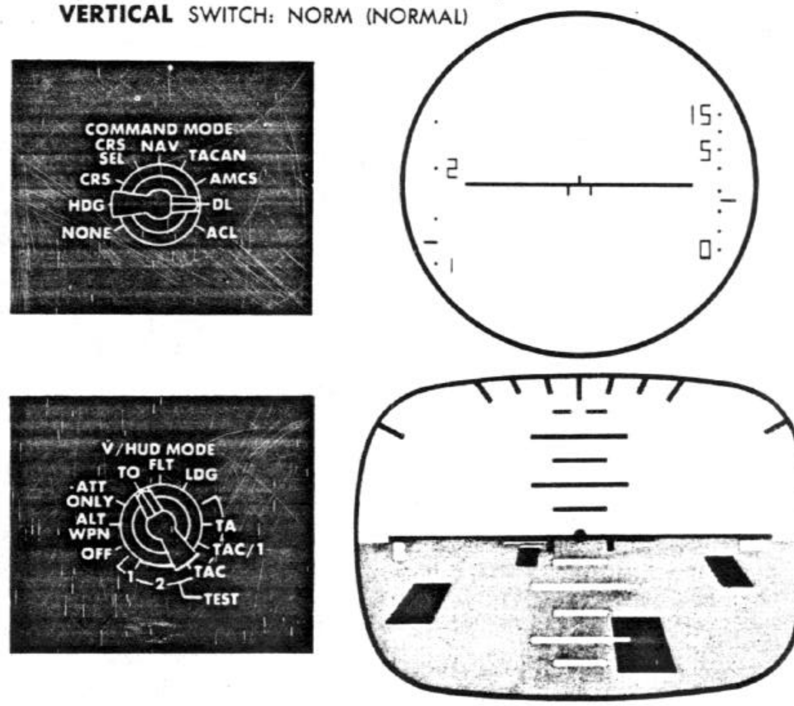
VDIG OPERATING MODES, Continued

COMMAND MODE SWITCH: D/L (DATA LINK)
V/HUD MODE SWITCH: T.O. (TAKEOFF)
VERTICAL SWITCH: NORM (NORMAL)

SYMBOL	DISPLAY DVI	DISPLAY PI
STEERING	X	X
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X	
COMMAND ALTITUDE ERROR	X	
TIME/RANGE	X**	X**
BREAKAWAY	X*	X*
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X	
ANGLE OF ATTACK		
RADAR ALTITUDE		X
TRUE AIRSPEED		X
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET		
INTERMEDIATE PITCH LINES	X	

NOTE

X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY DATA LINK
X** = PRESENT WHEN DATA LINK SENDS VALID SIGNAL



COMMAND MODE SWITCH: D/L (DATA LINK)
V/HUD MODE SWITCH: FLT (FLIGHT)
VERTICAL SWITCH: NORM (NORMAL)

SYMBOL	DISPLAY DVI	DISPLAY PI
STEERING	X	X
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X	X
COMMAND ALTITUDE ERROR	X	X
TIME/RANGE	X**	X**
BREAKAWAY	X*	X*
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X	
ANGLE OF ATTACK		
RADAR ALTITUDE		
TRUE AIRSPEED		
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET	X	
INTERMEDIATE PITCH LINES	X	

NOTE

X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY DATA LINK
X** = PRESENT WHEN DATA LINK SENDS VALID SIGNAL

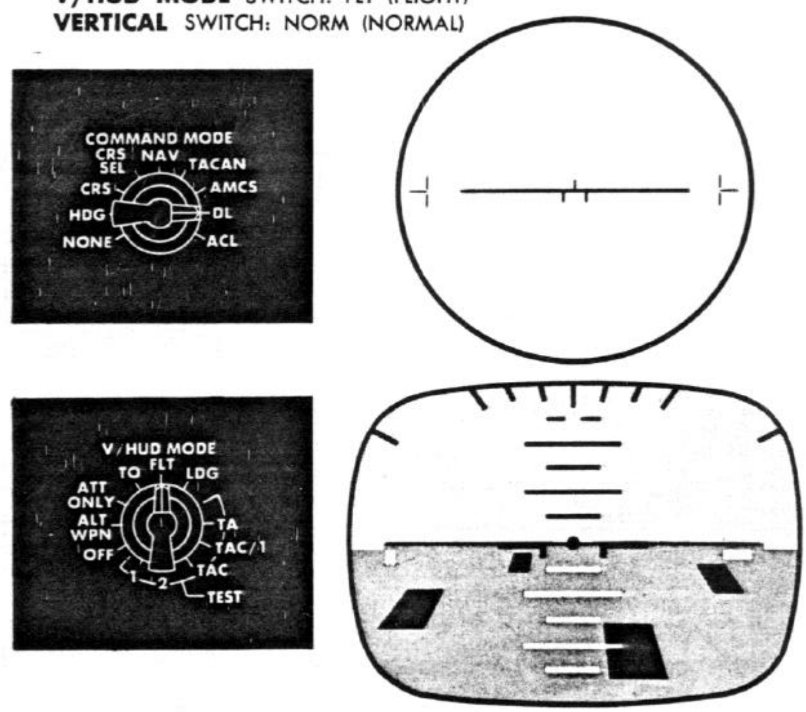


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Figure 8-6. (Sheet 3)

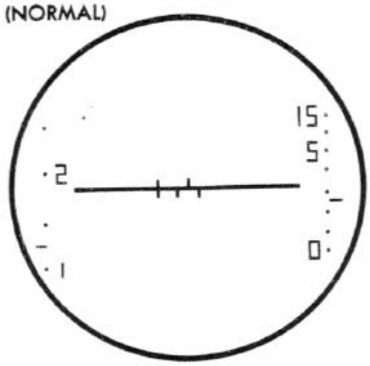
VDIG OPERATING MODES Continued

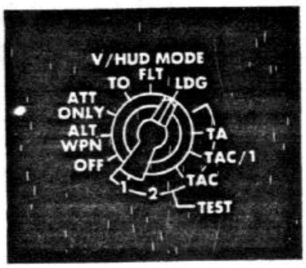
COMMAND MODE SWITCH: D/L (DATA LINK)
V/HUD MODE SWITCH: LDG (LANDING)
VERTICAL SWITCH: NORM (NORMAL)

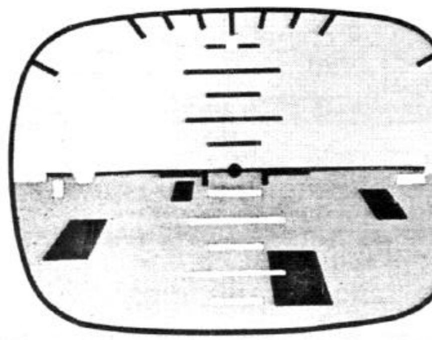
SYMBOL	DISPLAY DVI	DISPLAY PI
STEERING	X	X
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X	
COMMAND ALTITUDE ERROR	X	
TIME/RANGE	X*	X*
BREAKAWAY	X*	X*
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X	
ANGLE OF ATTACK	X**	X
RADAR ALTITUDE		X
TRUE AIRSPEED		X
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET		
INTERMEDIATE PITCH LINES	X	

NOTE

X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY DATA LINK
X** = ANGLE OF ATTACK BLINKS WHEN APPROACHING STALL





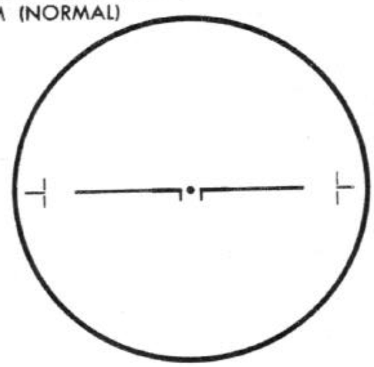


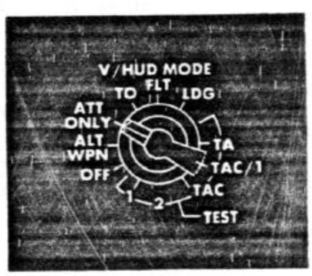
COMMAND MODE SWITCH: ANY SETTING BUT ACL
V/HUD MODE SWITCH: ATT ONLY (ATTITUDE ONLY)
VERTICAL SWITCH: NORM (NORMAL)

SYMBOL	DISPLAY DVI	DISPLAY PI
STEERING		
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X*	X*
COMMAND ALTITUDE ERROR	X*	X*
TIME/RANGE	X*	X*
BREAKAWAY	X*	X*
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X***	
ANGLE OF ATTACK		
RADAR ALTITUDE		
TRUE AIRSPEED		
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET	X**	
INTERMEDIATE PITCH LINES	X	

NOTE

X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY OTHER AIRCRAFT SUBSYSTEM EQUIPMENTS.
X** = ONLY WHEN COMMANDED BY AMCS
X*** = ALWAYS ON, HOWEVER IT DOES NOT MOVE





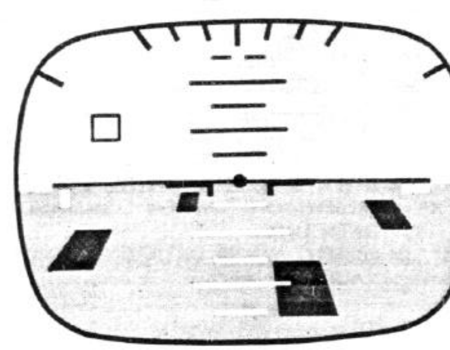


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Figure 8-6. (Sheet 4)

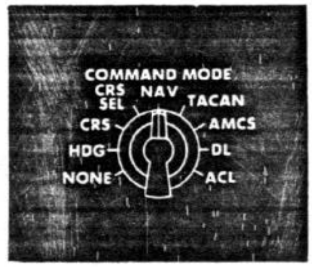
VDIG OPERATING MODES Continued

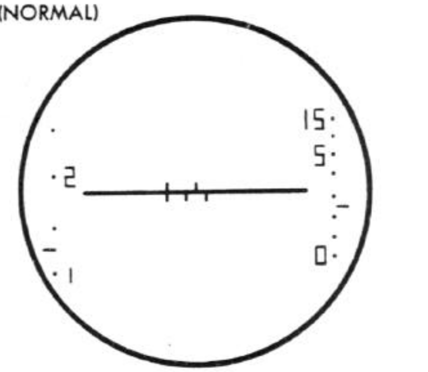
SYMBOL	DISPLAY DVI	PI
STEERING	X	X
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X*	
COMMAND ALTITUDE ERROR	X*	
TIME/RANGE		
BREAKAWAY		
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X	
ANGLE OF ATTACK	X**	X
RADAR ALTITUDE		X
TRUE AIRSPEED		X
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET		
INTERMEDIATE PITCH LINES	X	

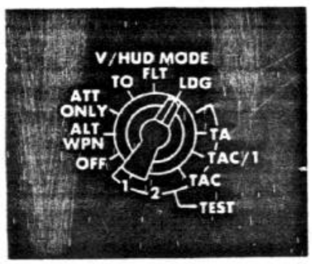
NOTE

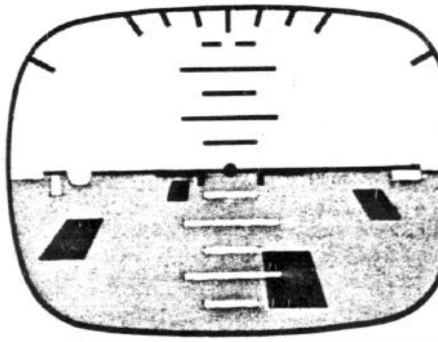
X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY OTHER AIRCRAFT SUBSYSTEM EQUIPMENTS.
▲ = HDG (HEADING), CRS (COURSE), CRS SEL (COURSE SELECT), NAV (NAVIGATION) OR TACAN
X** = ANGLE OF ATTACK BLINKS WHEN APPROACHING STALL

COMMAND MODE SWITCH: ANY NAVIGATION MODE ▲
V/HUD MODE SWITCH: LDG (LANDING)
VERTICAL SWITCH: NORM (NORMAL)







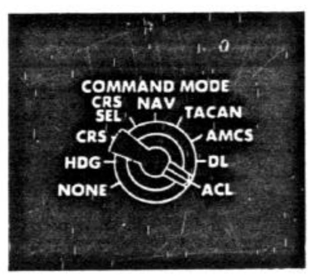


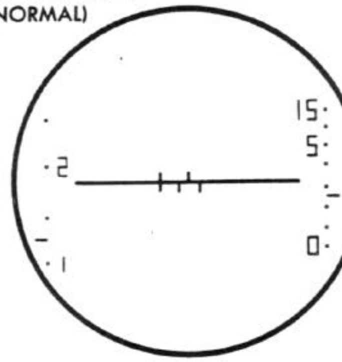
SYMBOL	DISPLAY DVI	PI
STEERING	X	X
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X*	
COMMAND ALTITUDE ERROR	X*	
TIME/RANGE		
BREAKAWAY	X***	X***
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X	
ANGLE OF ATTACK	X**	X
RADAR ALTITUDE		X
TRUE AIRSPEED		X
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET		
INTERMEDIATE PITCH LINES	X	

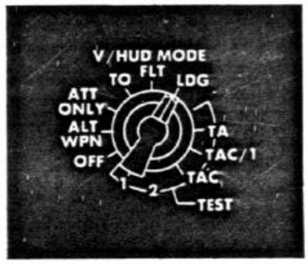
NOTE

X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY OTHER AIRCRAFT SUBSYSTEM EQUIPMENTS.
X** = ANGLE OF ATTACK BLINKS WHEN APPROACHING STALL
X*** = PRESENT ONLY WHEN DATA LINK SENDS MISSED MESSAGE OR WAVEOFF

COMMAND MODE SWITCH: ACL (AUTOMATIC CARRIER LANDING)
V/HUD MODE SWITCH: LDG (LANDING)
VERTICAL SWITCH: NORM (NORMAL)







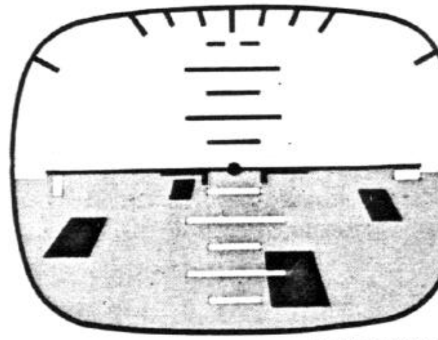


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Figure 8-6. (Sheet 5)

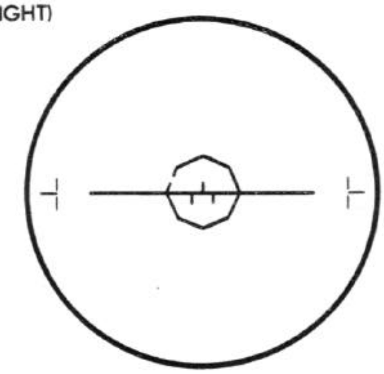
VDIG OPERATING MODES Continued

SYMBOL	DISPLAY DVI	DISPLAY PI
STEERING	X***	X***
MOVEABLE RETICLE		
AIRCRAFT RETICLE	X	X
COMMAND SPEED ERROR	X*	X*
COMMAND ALTITUDE ERROR	X*	X*
TIME/RANGE	X**	X***
BREAKAWAY	X*	X*
FIXED RETICLE		
HORIZON LINE	X	X
MAJOR PITCH LINES	X	X
ROLL POINTER	X	
REFERENCE CURSOR	X	
GROUND TEXTURE	X	
ANGLE OF ATTACK		
PRECISION COURSE VECTOR		
IMPACT POINT		
TARGET	X****	
INTERMEDIATE PITCH LINES	X	

NOTE

X = ALWAYS PRESENT ON DISPLAY
X* = PRESENT ONLY WHEN COMMANDED BY OTHER AIRCRAFT SUBSYSTEM EQUIPMENTS.
X** = ONLY ABOVE SYMBOLS WHEN IN NORM POSITION. WHEN IN BOTH POSITION, TACTICAL INFORMATION DISPLAY (TID) IS INTEGRATED WITH SOME OF ABOVE. WHEN IN TDC POSITION, ONLY TID DISPLAY APPEARS ON VERTICAL DISPLAY.
X***= WHEN AMCS SENDS PILOT DATA VALID.
X****=WHEN AMCS SENDS PILOT ABLE.

COMMAND MODE SWITCH: AMCS (AIRBORNE MISSILE CONTROL SYSTEM)
V/HUD MODE SWITCH: FLT (FLIGHT)
VERTICAL SWITCH: **



AMCS/FLT ALTERNATE SHOWING TIME AND RANGE AND TARGET SYMBOLS

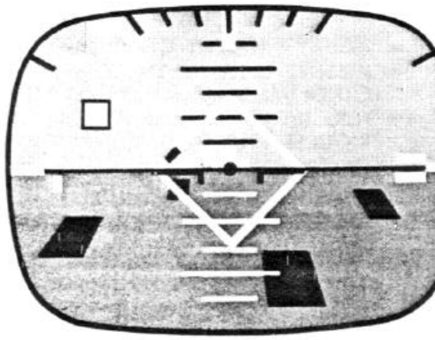


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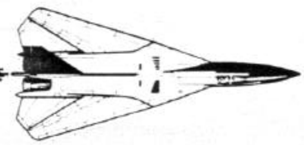
Figure 8-6. (Sheet 6)

section IX

FLIGHT CREW COORDINATION



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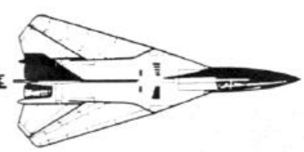
TO BE SUPPLIED AT A LATER DATE

section X

NATOPS EVALUATION



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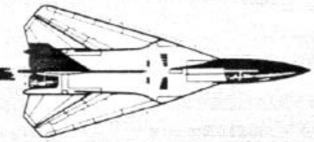
TO BE SUPPLIED AT A LATER DATE

section XI
PERFORMANCE DATA



part 1
INTRODUCTION

part 2
TAKE-OFF

part 3
CLIMB *

part 4
RANGE *

part 5
LANDING

*SEE SUPPLEMENTAL NATOPS FLIGHT MANUAL NAVAIR 01-10FAB-1A

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part 1

INTRODUCTION

INTRODUCTION

This section predicts the performance capabilities of the aircraft and serves as an aid in preflight planning. It is applicable to aircraft equipped with TF30-P-12 engines. The manual covers a flight spectrum ranging from sea level to approximately 60,000 feet and mach numbers from static to 2.5. All charts presented in the performance section of this manual are based on JP-5 fuel with a fuel density of 6.8 pounds per gallon.

This section is divided into five parts, with performance data presented in proper order for flight planning. Since the aircraft incorporates a variable sweep wing, there must, of necessity, be a repetition of data for a series of sweeps. This section covers a representative range of sweep angles from a minimum of 16 degrees to the maximum of 72.5 degrees. It should be noted that limiting conditions relative to altitude, airspeed, sweep, and CG control imposed on initial flights are not presented in this section.

"Operating Limitations", Section I of the basic Flight Manual, must be consulted for limiting conditions prior to all flight planning. All data are based on ARDC standard atmosphere (1959) and true mach number unless otherwise indicated. Pressure altitude is used in all data. The airspeed indicated on the airspeed mach indicator has been calibrated for pitot-static system errors by the CADC and therefore is actually KCAS (knots calibrated airspeed). However, this airspeed is referred to as KIAS (knots indicated airspeed) throughout this manual since it is read directly from the instrument. The symbols and abbreviations used throughout the section are defined as follows:

ABBREVIATIONS

A/B	Afterburner
Alt	Altitude
AMI	Airspeed Mach Indicator
AYC	Adverse Yaw Compensation
C°	Centigrade
CAS	Calibrated Airspeed
CG	Center of Gravity
EPR	Engine Pressure Ratio
Ft/Min	Feet per Minute
Ft/Sec	Feet per Second
F°	Fahrenheit
H_i	Indicated Altitude
IAS	Indicated Airspeed
ICAO	International Civial Aviation Organization
K	Airspeed Knots
KCAS	Knots Calibrated Airspeed
KTAS	Knots True Airspeed
KIAS	Knots Indicated Airspeed
δ LES	Degrees of leading edge slat deflection
LSTC	Low speed trim compensation
M	Mach number
M_t	True Mach number
Max	Maximum
MAX A/B	Maximum Afterburner
MIN A/B	Minimum Afterburner
MAC	Mean Aerodynamic Chord

ABBREVIATIONS

NM	Nautical Miles
OAT	Outside Air Temperature
PA or H_p	Pressure Altitude
PEC	Position Error Correction
PSI	Pounds per Square Inch
RPM	Revolutions per Minute
R/C	Rate of Climb
SL	Sea Level
Sweep ($\wedge$LE)	Angle of wing leading edge sweep relative to line perpendicular to fuselage center line.
TAS	True Airspeed
TE	Trailing Edge
δTEF	Degrees of trailing edge flap deflection
Temp	Temperature
T. O.	Takeoff
V_S OR V_{SL}	Stall speed, power off
V_{SPA}	Stall speed, approach power
ΔM	Mach number increment
ΔH	Altitude increment
ΔV	Airspeed increment
δ	Delta: ratio of ambient air pressure to standard sea level air pressure
σ	Sigma: ratio of ambient air density to standard sea level air density

POSITION ERROR CORRECTION

Total and static pressures sensed by the pitot-static probe on the nose boom provide altitude and airspeed data for the flight instruments. Because of the influence of aircraft on the flow field around the static pressure parts on pitot-static probe at some conditions, an error is introduced into the system. This error, termed position error, is caused by static pressure ports measuring a small component of dynamic pressure as well as existing static pressure. Figures 11-1 and 11-2 provide charts for the determination of this position error for airspeed, altitude and mach instruments. Figure 11-1 presents data for converting indicated airspeed (KIAS) to calibrated airspeed (KCAS) and indicated altitudes to true altitudes. This figure may also be used in reverse, i.e., for converting calibrated airspeeds to indicated airspeeds and true pressure altitudes to indicated pressure altitudes. At mach numbers greater than 1.05, no position error exists.

Sample Problem

Given:

ALT = 30,000 feet
CAS = 260 KTS

Find:

The position error correction, the airspeed and altimeter readings.

Solution:

Refer to figure 11-1 and note: to use the chart, the given value 260 KCAS must be converted to a corrected indicated airspeed, i.e., the chart for this problem must be used in reverse. To use the chart in reverse, an iterative procedure is involved, which is as follows: Enter the chart at indicated airspeed scale with 260 KCAS value and project vertically to intersect an indicated altitude of 30,000 feet from which read a ΔV_{PEC} of 8 knots.

Substituting the known data into the rearranged given equation on the chart, obtain:

$$V_i = V_c - \Delta V_{PEC}$$

$$V_i = 260 - 8 = 252 \text{ KIAS}$$

Reenter figure 11-1 at the computed indicated airspeed of 252 and obtain a new ΔV_{PEC} of 7.8 knots which, for all purposes, is 8 knots as previously obtained. Due to this small change in ΔV_{PEC}, no further iteration is necessary and the value of 252 can be used as the indicated airspeed for this problem. Using this speed, 252 KIAS, follow the chase around lines and obtain a ΔV_{PEC} of 8 knots and ΔH_{PEC} of 500 feet. This indicated pressure altitude for the problem is then obtained through the use of the rearranged altitude equation.

$$H_{pi} = H_p - \Delta H_{PEC}$$

$$H_{pi} = 30,000 - 500 = 29,500 \text{ feet.}$$

An example for the use of the position error correction, figure 11-2, to the mach instrument is not presented since it is assemed that the use of the chart is obvious.

AIRSPEED - MACH NUMBER

The curves shown in figure 11-3 (sheets 1, 2, and 3) are presented an aid in conversion between calibrated airspeed, true airspeed, and mach number at various altitude and temperature conditions. On

the chart is shown an example for its use. The problem and solution are outlined as follows:

Sample Problem

Given:

CAS = 300 KTS
ALT = 25,000 feet
OAT = 20° C

Find:

Mach number and TAS

Solution:

Enter figure 11-3, sheet 1 at (A) and 300 KTS and proceed vertically to 25,000 feet to (B), then horizontally to the left to read M = .717 (C), and to the right to the sea level base line and down to 20°C (D), and the across to read the TAS of 479 KTS at (E). Note on figure 11-3, sheet 3, the base line is 30,000 feet rather than sea level.

MISCELLANEOUS CHARTS

Charts 11-4 through 11-7 are presented for general information and are felt to be self-explanatory without the necessity of examples of their use. Instructions for use are shown on charts as required.

TAKE-OFF AND LANDING WIND COMPONENTS CHART

A standard take-off and landing wind components chart (figure 11-8) is presented for computation of crosswind and headwind components. Crosswind directions are presented from 0 to 90 degrees in 10 degree increments and windspeeds from 0 to 60 knots in 1 knot increments.

Sample Problem

Given:

Windspeed - 35 KTS

Wind direction - 050 degrees

Runway Heading - 030 degrees

Find:

Crosswind component

Headwind component

Solution:

Reduce the wind direction to a relative bearing by determining the difference between wind direction and runway heading. Enter the chart with a relative bearing of 20 degrees. Move along the relative bearing angle to intercept the windspeed arc of 35 knots at point (A). From this point, move vertically downward to read a crosswind component of 12 knots at point (B). To find the headwind component, move horizontally to the left from point (A) to read a headwind of 33 knots at point (C).

POSITION ERROR CORRECTION

AIRCRAFT CONFIGURATION
NO EXTERNAL STORES
IR HEAD INSTALLED

DATE: 15 MARCH 1968
DATA BASIS: FLIGHT TEST

REMARKS
ENGINE(S): (2) TF30-P-12
TEST NOSE BOOM
NACA A-6 PITOT TUBE
CALIBRATION T-3

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

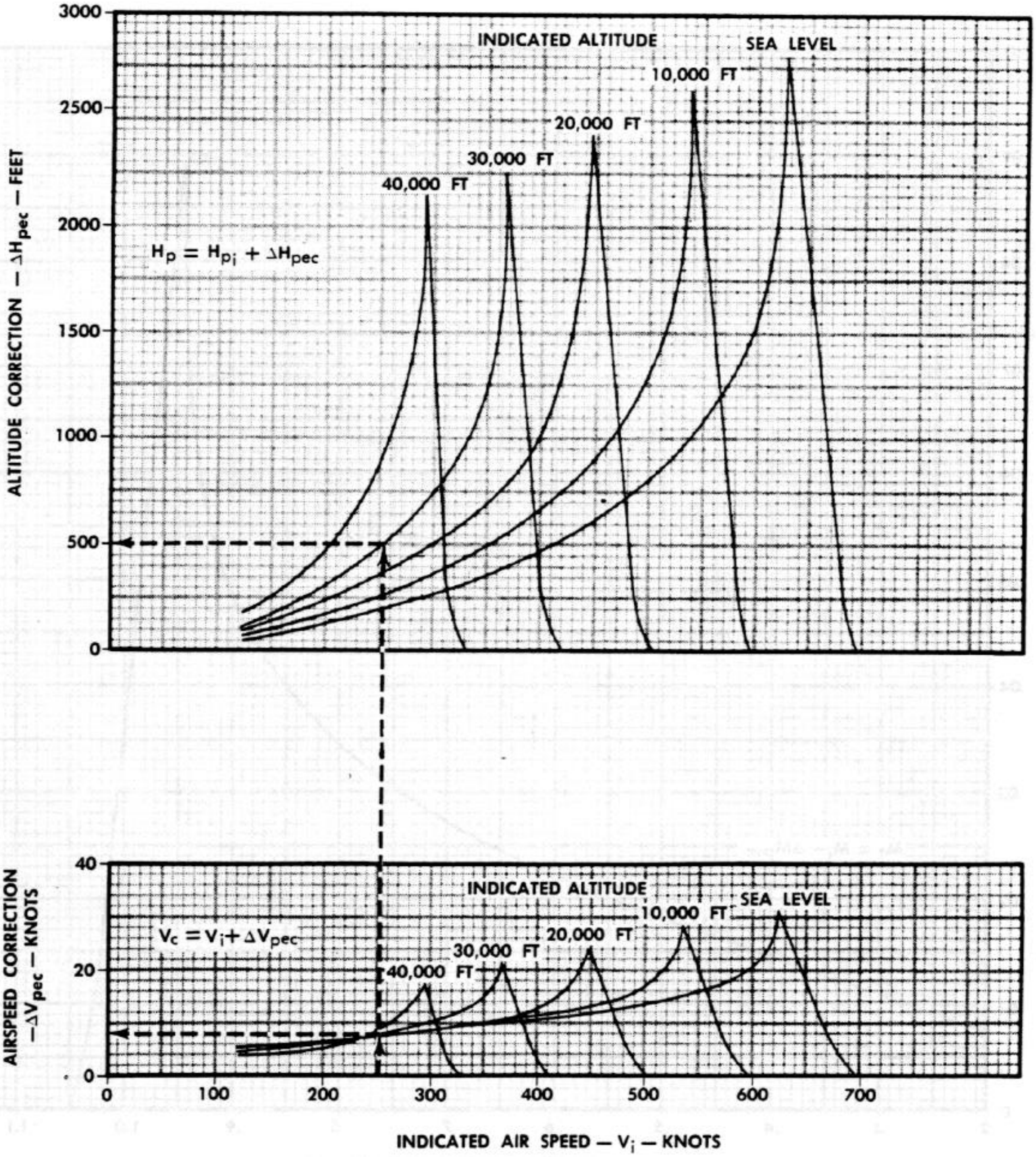

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Figure 11-1

POSITION ERROR CORRECTION

AIRCRAFT CONFIGURATION
NO EXTERNAL STORES
IR HEAD INSTALLED

DATE: 15 MARCH 1968
DATA BASIS: FLIGHT TEST

REMARKS
ENGINE(S): (2) TF30-P12
TEST NOSE BOOM
NACA A-6 PITOT TUBE
CALIBRATION T-3

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

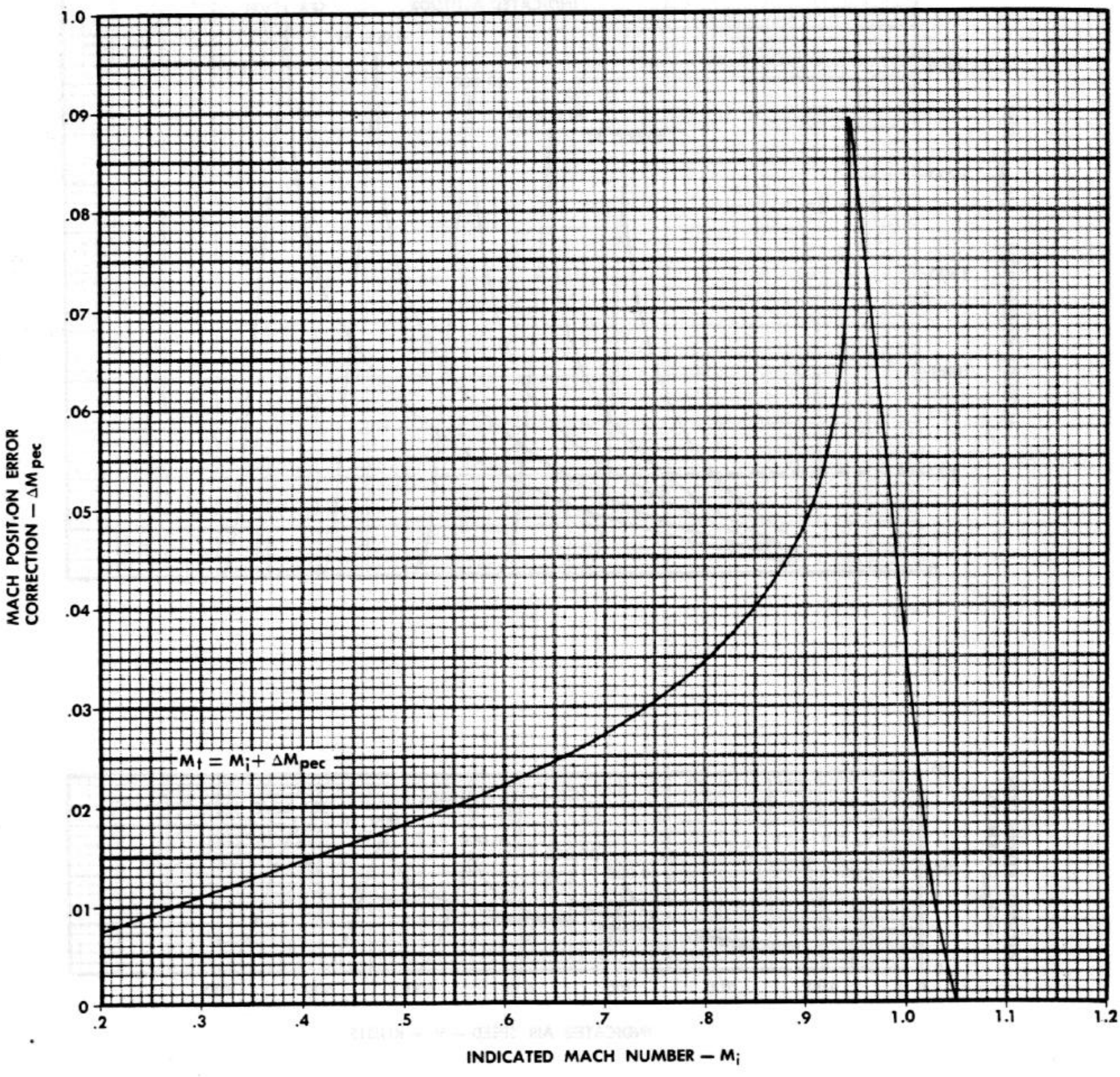

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Figure 11-2

AIRSPEED MACH NUMBER

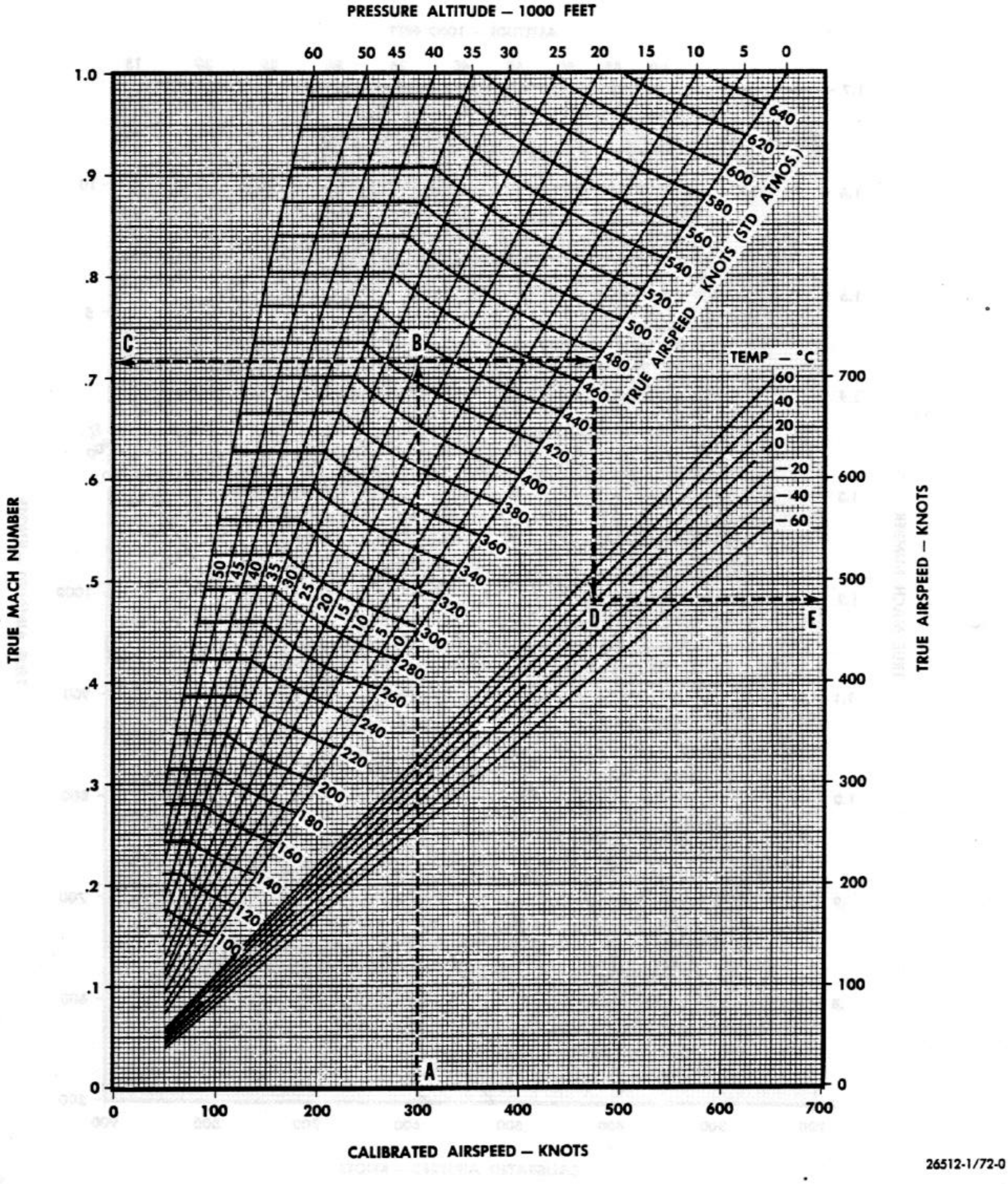

Figure 11-3 (Sheet 1)

AIRSPEED MACH NUMBER

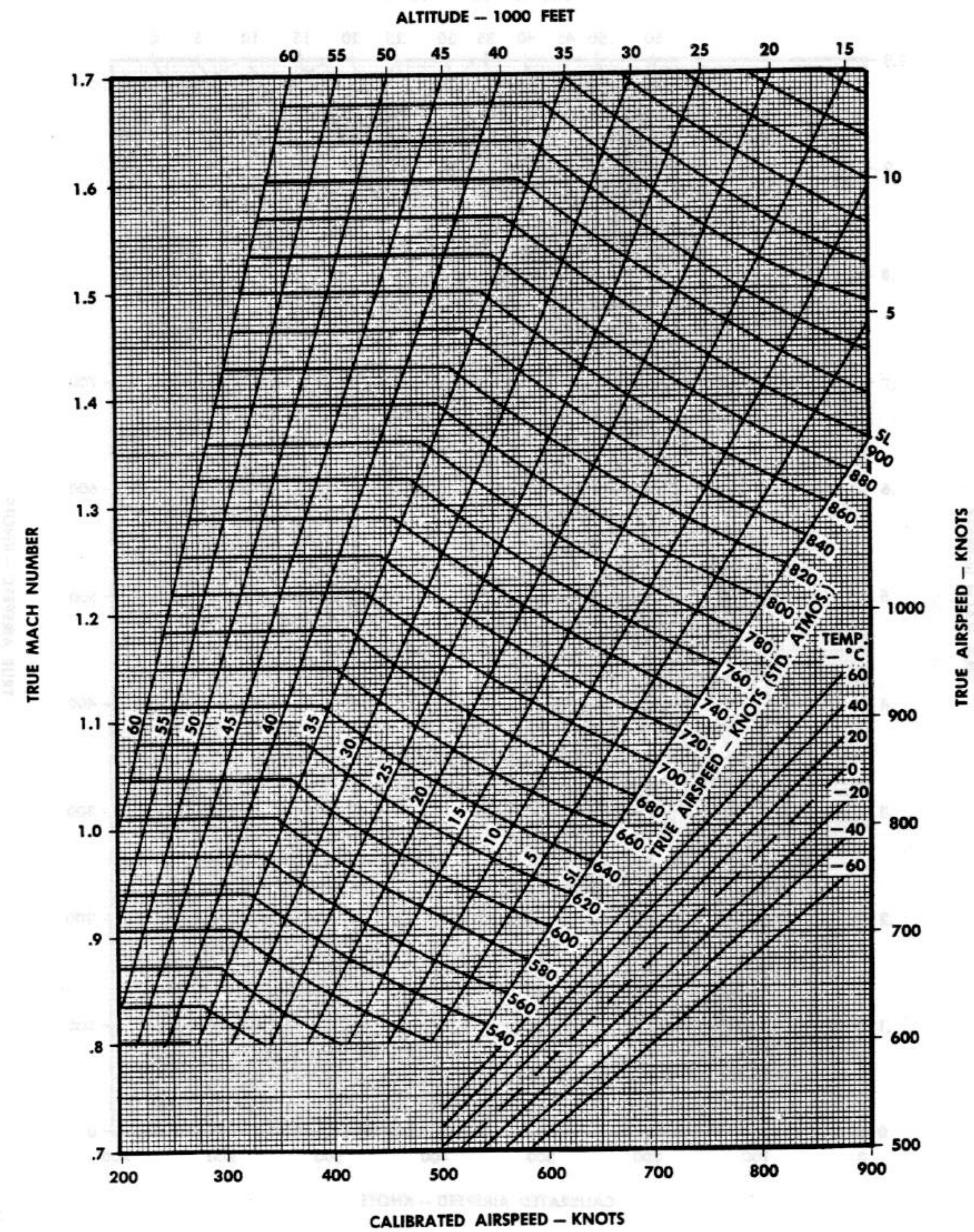

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Figure 11-3 (Sheet 2)

AIRSPEED MACH NUMBER

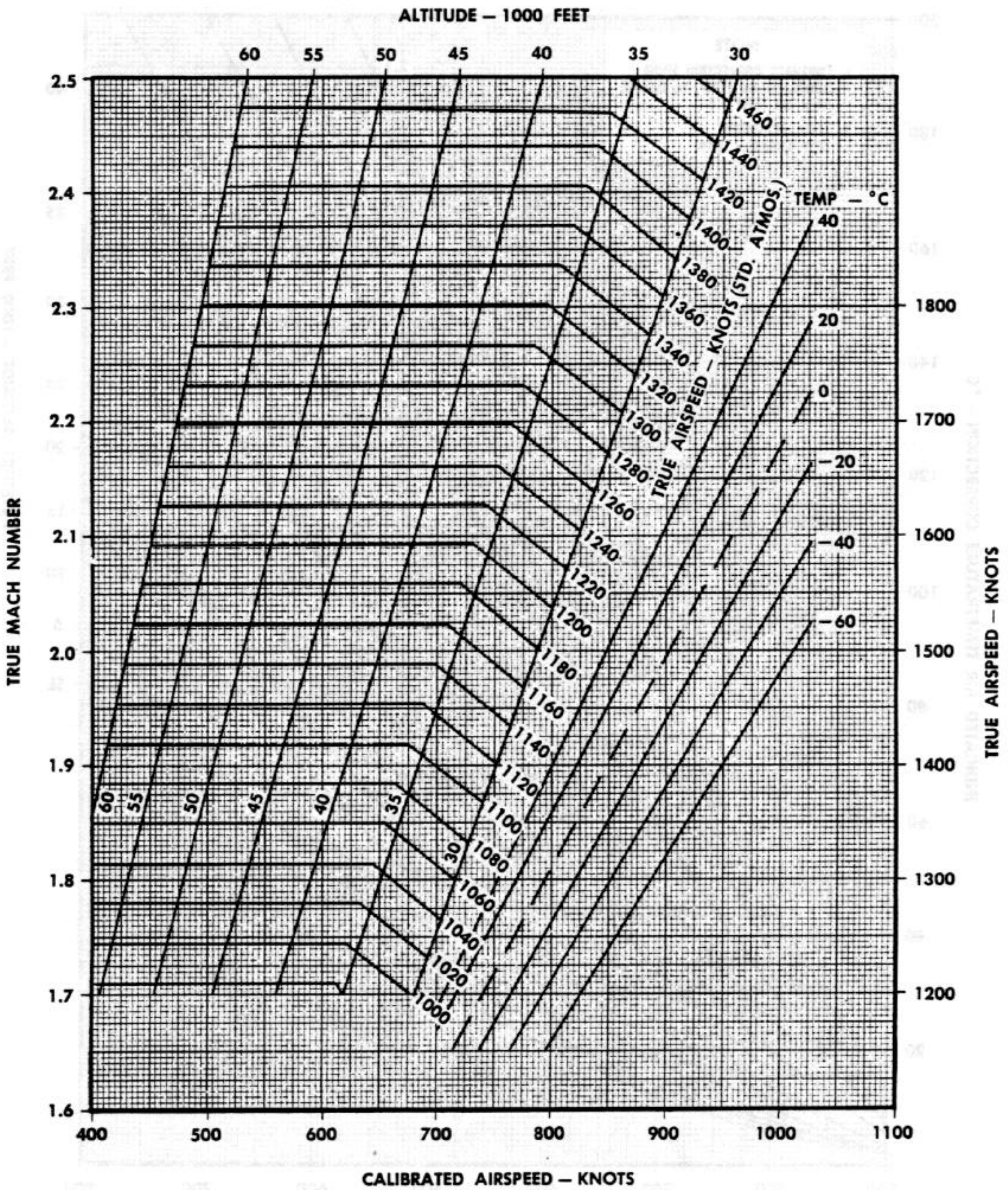

Figure 11-3 (Sheet 3)

TEMPERATURE CORRECTION FOR COMPRESSIBILITY

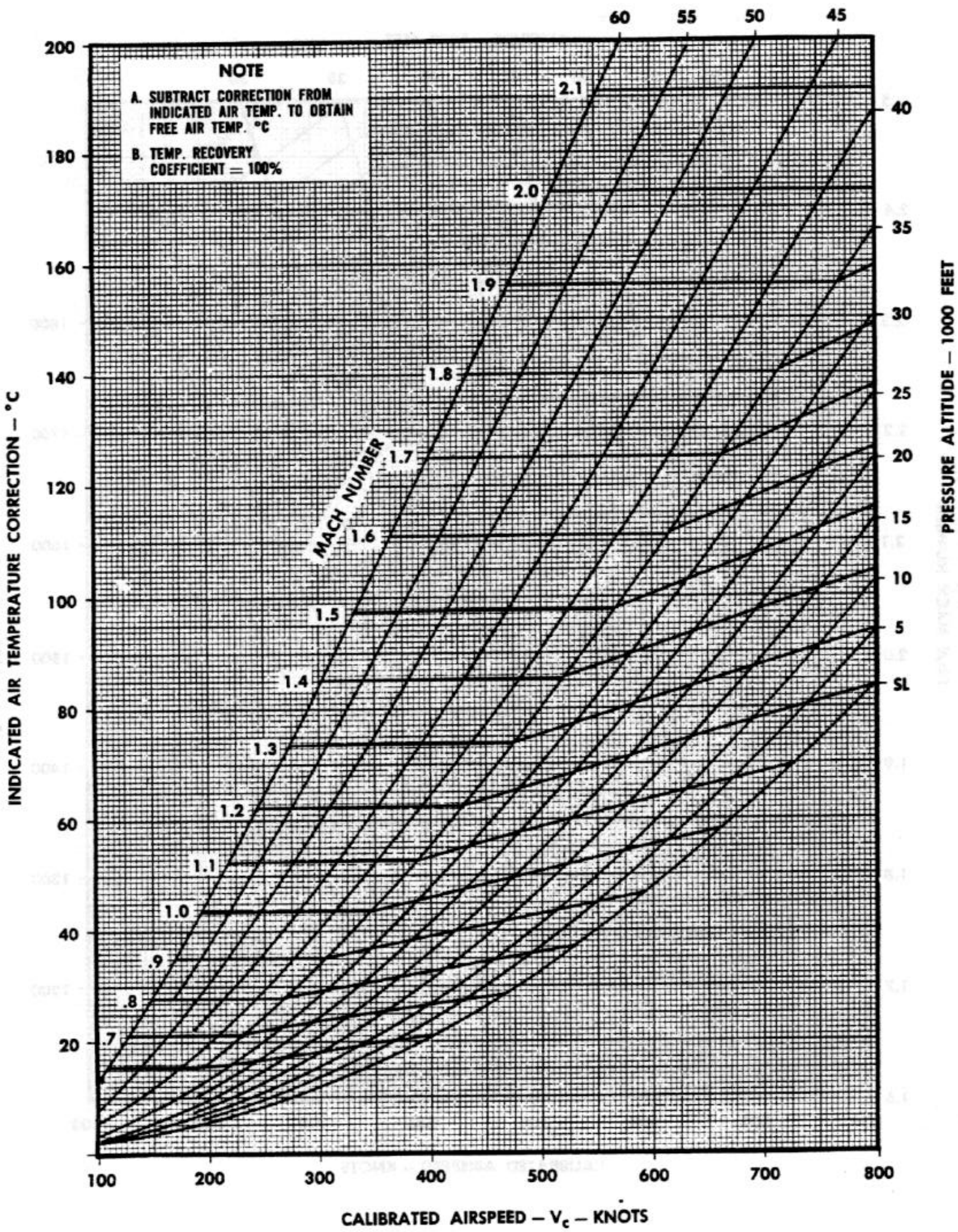

Figure 11-5 (Sheet 1)

TEMPERATURE CORRECTION FOR COMPRESSIBILITY

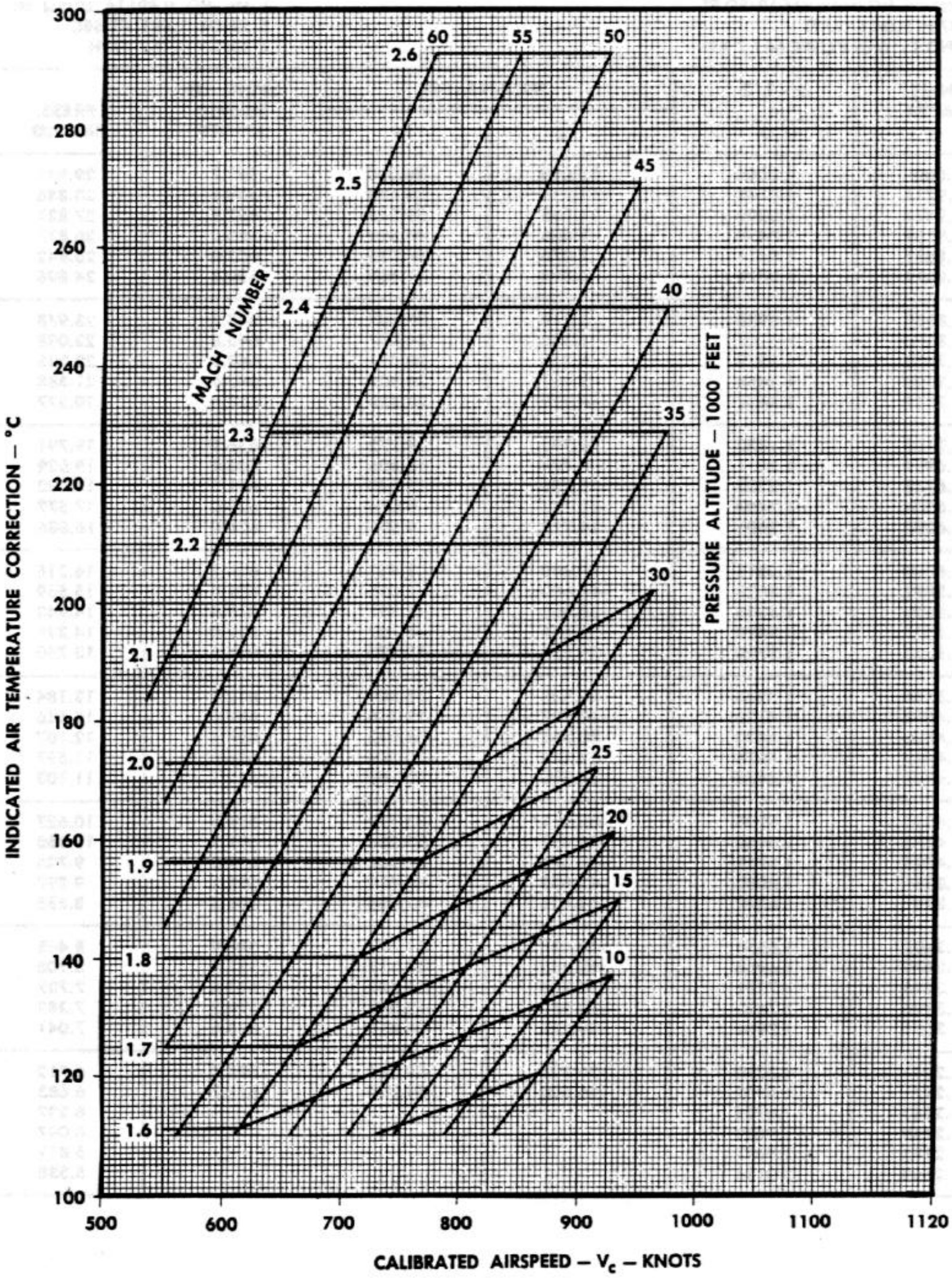

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Figure 11-5 (Sheet 2)

STANDARD ATMOSPHERE TABLE

STANDARD SL CONDITIONS

TEMPERATURE 15°C (59°F)
PRESSURE 29.921 IN. HG 2116.216 LB/SQ FT
DENSITY .0023769 SLUGS/CU FT
SPEED OF SOUND 1116.89 FT/SEC 661.7 KTS

CONVERSION FACTORS

1 IN. HG 70.727 LB/SQ FT
1 IN. HG 0.49116 LB/SQ IN.
1 KNOT 1.688 FT/SEC
1 KNOT 1.151 MPH

ATTITUDE FEET	DENSITY RATIO σ	σ- 1/2 $\frac{1}{\sqrt{\sigma}}$	TEMPERATURE °C	TEMPERATURE °F	SPEED OF SOUND KNOTS	PRESS. IN. HG	PRESS. RATIO δ
0	1.000	1.0000	15.000	59.000	661.7	29.921	1.0000
1000	.9711	1.0148	13.019	55.434	659.5	28.856	.9644
2000	.9428	1.0299	11.038	51.868	657.2	27.821	.9298
3000	.9151	1.0454	9.056	48.302	654.9	26.817	.8962
4000	.8881	1.0611	7.076	44.735	652.6	25.842	.8637
5000	.8617	1.0773	5.094	41.169	650.3	24.896	.8320
6000	.8359	1.0938	3.113	37.603	648.7	23.978	.8014
7000	.8106	1.1107	1.132	34.037	645.6	23.088	.7716
8000	.7860	1.1279	-0.850	30.471	643.3	22.225	.7428
9000	.7620	1.1456	-2.831	26.905	640.9	21.388	.7148
10,000	.7385	1.1637	-4.812	23.338	638.6	20.577	.6877
11,000	.7155	1.1822	-6.793	19.772	636.2	19.791	.6614
12,000	.6932	1.2011	-8.774	16.206	633.9	19.029	.6360
13,000	.6713	1.2205	-10.756	12.640	631.5	18.292	.6113
14,000	.6500	1.2403	-12.737	9.074	629.0	17.577	.5875
15,000	.6292	1.2606	-14.718	5.508	626.0	16.886	.5643
16,000	.6090	1.2815	-16.699	1.941	624.2	16.216	.5420
17,000	.5892	1.3028	-18.680	-1.625	621.8	15.569	.5203
18,000	.5699	1.3246	-20.662	-5.191	619.4	14.942	.4994
19,000	.5511	1.3470	-22.643	-8.757	617.0	14.336	.4791
20,000	.5328	1.3700	-24.624	-12.323	614.6	13.750	.4595
21,000	.5150	1.3935	-26.605	-15.889	612.1	13.184	.4406
22,000	.4976	1.4176	-28.587	-19.456	609.6	12.636	.4223
23,000	.4806	1.4424	-30.568	-23.022	607.1	12.107	.4046
24,000	.4642	1.4678	-32.549	-26.588	604.6	11.597	.3876
25,000	.4481	1.4938	-34.530	-30.154	602.1	11.103	.3711
26,000	.4325	1.5206	-36.511	-33.720	599.6	10.627	.3552
27,000	.4173	1.5480	-38.492	-37.286	597.1	10.168	.3398
28,000	.4025	1.5762	-40.474	-40.852	594.6	9.725	.3250
29,000	.3881	1.6052	-42.455	-44.419	592.1	9.297	.3107
30,000	.3741	1.6349	-44.436	-47.985	589.5	8.885	.2970
31,000	.3605	1.6654	-46.417	-51.551	586.9	8.488	.2837
32,000	.3473	1.6968	-48.398	-55.117	584.4	8.106	.2709
33,000	.3345	1.7291	-50.379	-58.683	581.8	7.737	.2586
34,000	.3220	1.7623	-52.361	-62.249	579.2	7.382	.2467
35,000	.3099	1.7964	-54.243	-65.816	576.6	7.041	.2353
36,000	.2981	1.8315	-56.323	-69.382	574.0	6.712	.2243
36,089	.2971	1.8347	-56.500	-69.700	573.7	6.683	.2234
37,000	.2843	1.8753				6.397	.2138
38,000	.2710	1.9209				6.097	.2038
39,000	.2583	1.9677				5.811	.1942
40,000	.2462	2.0155				5.538	.1851

26512-1/80.1-0

Figure 11-6 (Sheet 1)

STANDARD ATMOSPHERE TABLE

STANDARD SL CONDITIONS

TEMPERATURE 15°C (59°F)
PRESSURE 29.921 IN. HG 2116.216 LB/SQ FT
DENSITY .0023769 SLUGS/CU FT
SPEED OF SOUND 1116.89 FT/SEC 661.7 KTS

CONVERSION FACTORS

1 IN. HG 70.727 LB/SQ FT
1 IN. HG 0.49116 LB/SQ IN.
1 KNOT 1.688 FT/SEC
1 KNOT 1.151 MPH

ATTITUDE FEET	DENSITY RATIO σ	σ - 1/2 $\frac{1}{\sqrt{\sigma}}$	TEMPERATURE		SPEED OF SOUND KNOTS	PRESS. IN. HG	PRESS. RATIO δ
			°C	°F			
41,000	.2346	2.0645	-56.500	-69.700	573.7	5.278	.1764
42,000	.2236	2.1148				5.030	.1681
43,000	.2131	2.1662				4.794	.1602
44,000	.2031	2.2189				4.569	.1527
45,000	.1936	2.2728				4.355	.1455
46,000	.1845	2.3281				4.151	.1387
47,000	.1758	2.3848				3.956	.1322
48,000	.1676	2.4428				3.770	.1260
49,000	.1597	2.5022				3.593	.1201
50,000	.1522	2.5630				3.425	.1145
51,000	.1451	2.6254				3.264	.1091
52,000	.1383	2.6892				3.111	.1040
53,000	.1318	2.7546				2.965	.09909
54,000	.1256	2.8216				2.826	.09444
55,000	.1197	2.8903				2.693	.09001
56,000	.1141	2.9606				2.567	.08578
57,000	.1087	3.0326				2.446	.08176
58,000	.1036	3.1063				2.331	.07792
59,000	.09877	3.1819				2.222	.07426
60,000	.09414	3.2593				2.118	.07078
61,000	.08972	3.3386	-56.500	-69.700	573.7	2.018	.06746
62,000	.08551	3.4198				1.924	.06429
63,000	.08150	3.5029				1.833	.06127
64,000	.07767	3.5881				1.747	.05840
65,000	.07403	3.6754				1.665	.05566
66,000	.07055	3.7649				1.587	.05305
67,000	.06724	3.8564				1.513	.05056
68,000	.06409	3.9502				1.442	.04819
69,000	.06108	4.0463				1.374	.04592
70,000	.05821	4.1447				1.310	.04377
71,000	.05548	4.2456				1.248	.04171
72,000	.05288	4.3488				1.190	.03976
73,000	.05040	4.4545				1.134	.03789
74,000	.04803	4.5633				1.081	.03611
75,000	.04578	4.6738				1.030	.03442
76,000	.04363	4.7874				0.982	.03280
77,000	.04158	4.9039				0.935	.03126
78,000	.03963	5.0231				0.892	.02980
79,000	.03777	5.1454				0.850	.02840
80,000	.03600	5.2706				0.810	.02707

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Figure 11-6 (Sheet 2)

DENSITY ALTITUDE

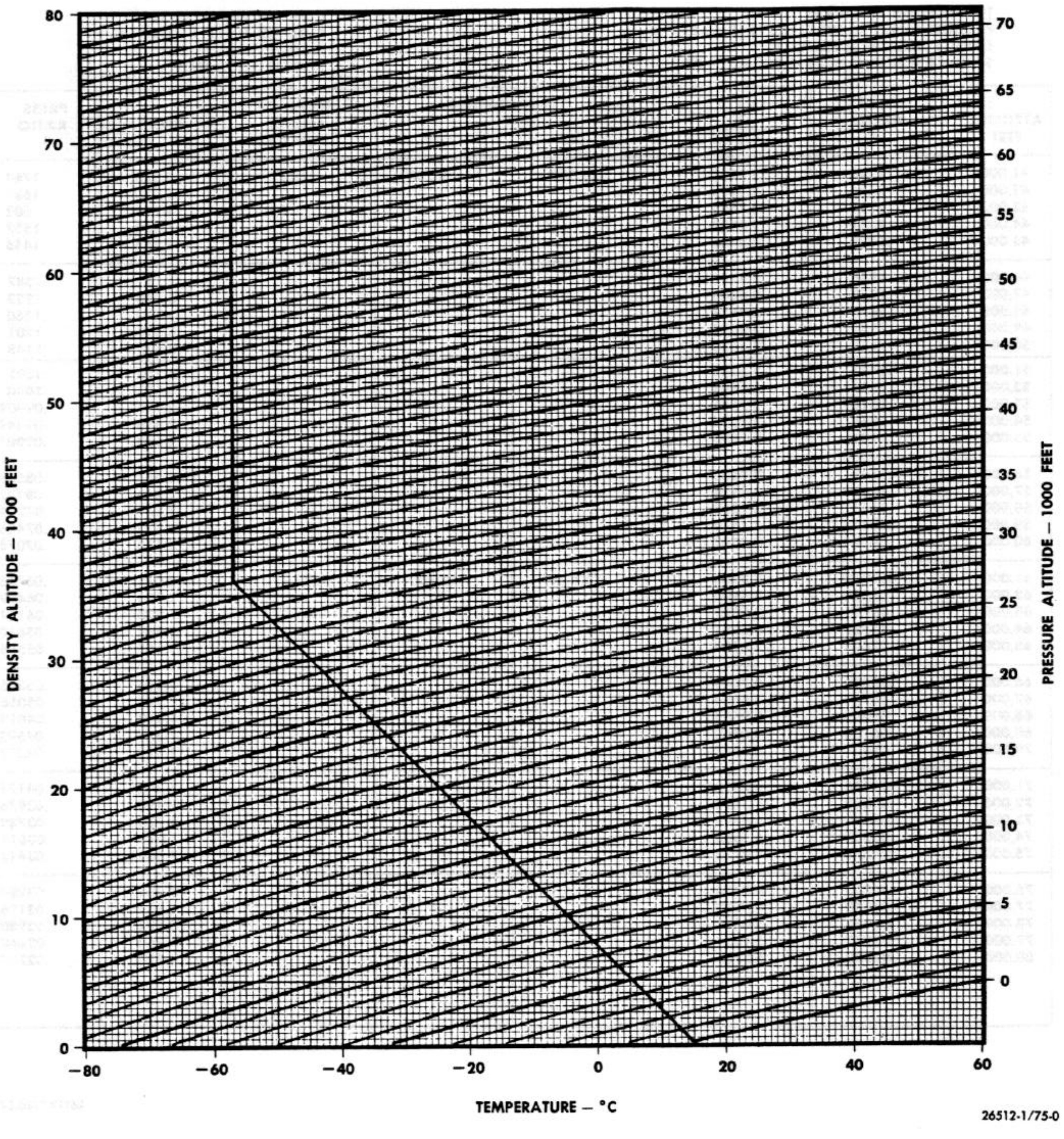

Figure 11-4

TEMPERATURE CONVERSION

SAMPLE PROBLEM

A. AIR TEMPERATURE 59°F
C. AIR TEMPERATURE 15°C

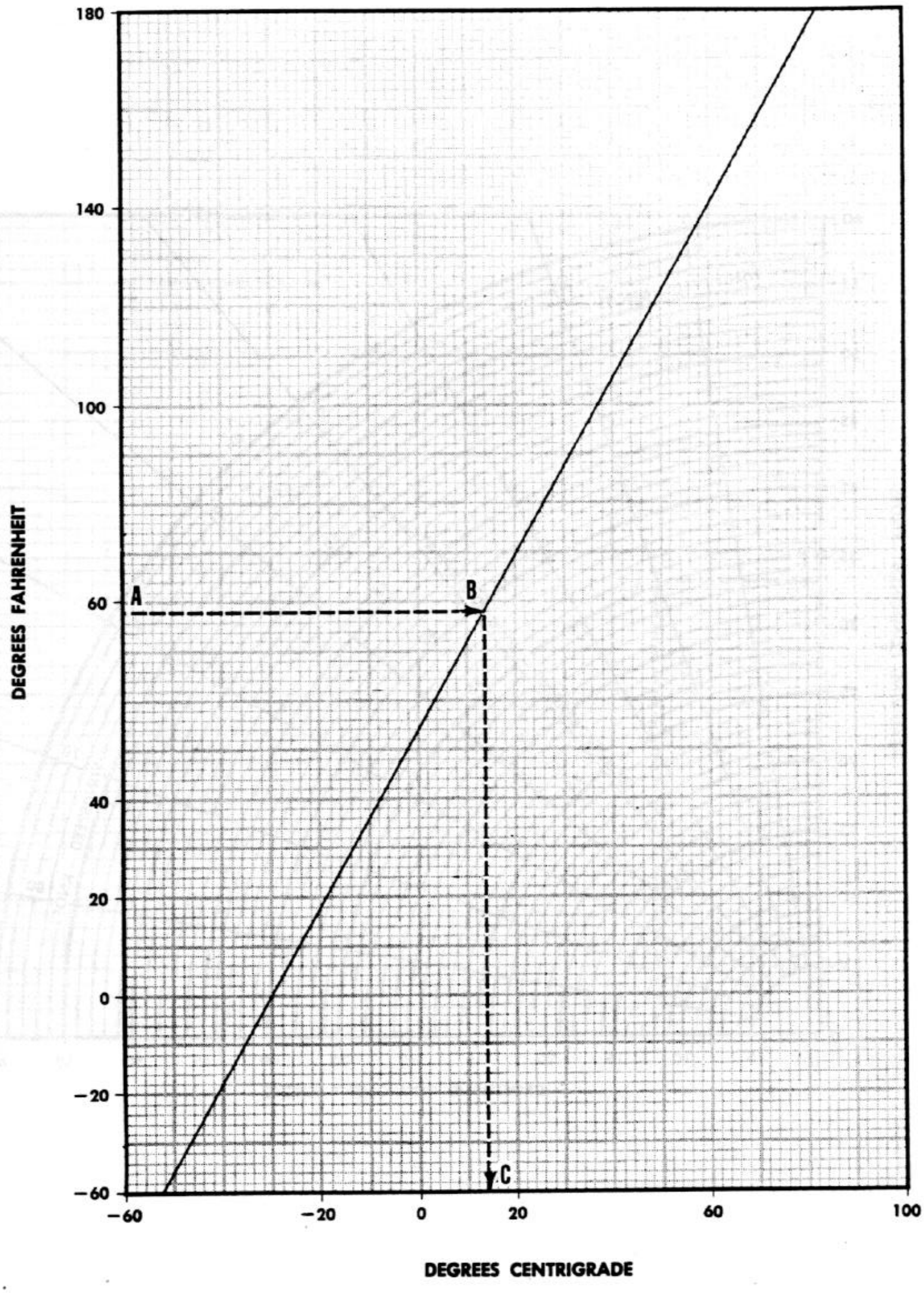

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Figure 11-7

TAKE-OFF AND LANDING WIND COMPONENTS CHART

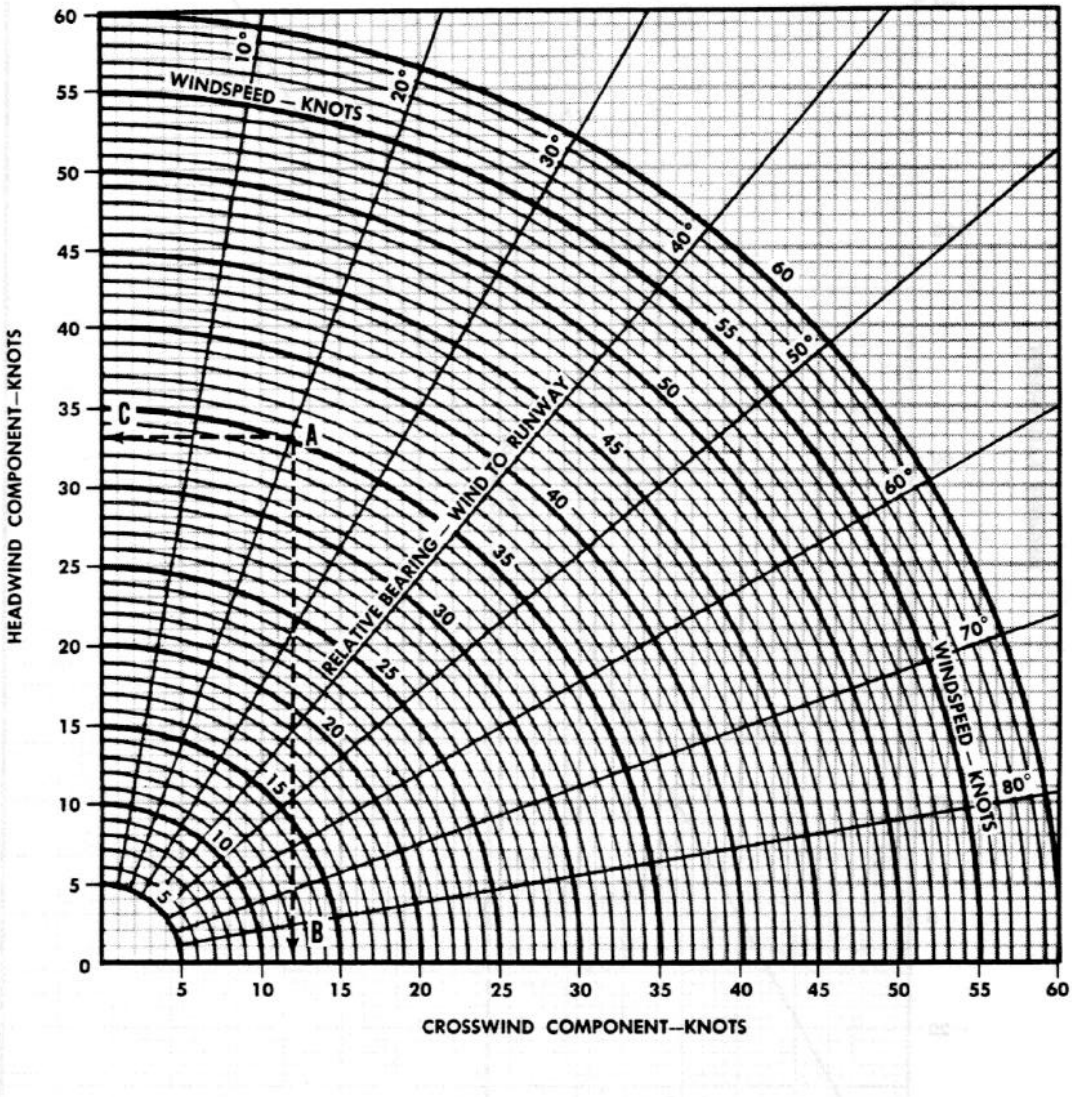

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Figure 11-8

part 2
TAKE-OFF

TABLE OF CONTENTS

TAKEOFF

This section presents data necessary to takeoff planning for a sweep angle of 16° and for a maximum afterburner thrust setting. The basic takeoff charts show distance in ground roll and total distance over an obstacle as affected by air temperature, field elevation, C.G. position, runway slope, gross weight and wind conditions. Airspeeds are shown for rotation from ground roll attitude to takeoff attitude, takeoff, and climb-out. For information relative to determination of center-of-gravity position, Section V of T.O. 1F-111A-1 must be consulted. Takeoff charts are presented for flap settings of 40° and 25° deflections. Other charts presented include, takeoff flap setting for single engine climb, critical engine failure and refusal speeds, critical field length, critical field length with center-of-gravity corrections, effect of runway conditions, continuation speeds, velocity during takeoff ground run, and stopping distance data.

DEFINITION OF TERMS

Takeoff Speed

Speed at which the main gear leaves the ground.

Takeoff Ground Run Distance

Ground run in feet from brake release to takeoff speed.

Rotation Speed

Speed at which the pilot starts to rotate the aircraft for takeoff.

Climb-out Speed

Speed at 50 feet above the runway and is speed to be used for climb-out.

Climb-out Distance

Horizontal distance from takeoff point to obstacle.

Go/No-Go Distance

The go/no-go distance is the line check marker distance from the start of the takeoff run to the first runway marker below refusal distance.

Go/No-Go Speed

Minimum speed at the go/no-go marker.

Acceleration Check Distance

Distance to the runway marker which is 2000 feet short of the go/no-go distance.

Acceleration Check Speed

Minimum speed at the acceleration check marker.

Critical Engine Failure Speed

The speed at which the aircraft, after an engine failure, will accelerate to liftoff in the same distance required to decelerate to a complete stop.

Continuation Speed (Decision Speed)

Minimum speed from which a safe takeoff can be continued in the remaining runway length should engine failure occur.

Critical Field Length

The runway length required to accelerate the aircraft to critical engine failure speed, experience an engine failure, and then either takeoff with the remaining engine or decelerate the aircraft to a complete stop.

Refusal Speed

Maximum speed that may be attained and still stop the aircraft at the end of the runway, should engine failure occur during takeoff.

Refusal Distance

The distance required to accelerate to refusal speed under normal conditions.

Tire Limit Speed

Maximum safe speed for ground run. Takeoff speed should not exceed tire limit speed.

Single Engine Acceleration

Acceleration with one engine operation at maximum afterburner thrust.

Braked Deceleration

Deceleration on runway aided by application of brakes.

Runway Condition Reading (RCR)

The number portion of a system of reporting surface conditions at terminal airfields, related to the effectiveness of braking on the runway.

Runway Slope

Expressed in percent (uphill or downhill). The runway slope is the change in runway height divided by the runway length multiplied by 100.

Runway Elevation

Altitude above sea level of the runway location.

OAT

Outside Air Temperature.

TAKEOFF/ABORT

The takeoff/abort criteria is illustrated in figure 11-8. The takeoff/abort charts contained in figures 11-17 thru 11-25 provide the means of planning for a GO-NO-GO decision should engine failure occur during takeoff. A general discussion of the GO-NO-GO concept is provided in this paragraph to illustrate the factors which influence the decision to stop or go if an engine failure occurs. The principal factor affecting an aborted takeoff is the relationship of actual runway length to critical field length. This relationship falls into three categories, and within each category, the speed at which engine failure occurs further affects the stop or go decision, as follows:

Category 1

Runway Length Greater Than Critical Field Length (refusal speed exceeds critical engine failure speed).

a. If engine failure occurs below critical engine failure speed, the aircraft should be stopped, as runway length will always be sufficient for stopping. Takeoff distance increases as engine failure speed decreases, and may exceed the runway length under certain conditions.

b. If engine failure occurs between critical engine failure speed and refusal speed, the aircraft can takeoff or stop within remaining distance.

c. If engine failure occurs above refusal speed, the aircraft should continue takeoff as it would overrun runway in stopping. Sufficient runway for takeoff will be available.

Category 2

Runway Length Same as Critical Field Length, (refusal speed and critical engine failure speed coincide).

a. Aircraft should be stopped if below, and should continue takeoff if above the coincidence speed. Runway will be adequate for either condition.

Category 3

Runway Length Less Than Critical Field Length (refusal speed less than critical engine failure speed). This is the most dangerous situation and takeoff is not recommended. If takeoff is imperative, the following conditions will exist.

a. If engine failure occurs before refusal speed. Aircraft should be stopped. Runway will always be sufficient for stopping.

b. If engine failure occurs between refusal speed and critical engine failure speed. Aircraft cannot stop or takeoff within the remaining runway.

c. If engine failure occurs above critical engine failure speed. Aircraft should continue attempted takeoff. If runway is much less than critical field length, takeoff may not be possible.

DATA BASIS FOR CHARTS

Takeoff Technique. (To 50 feet above the runway)

At rotation speed apply back pressure to the stick to achieve a moderate rotation rate that will result in takeoff eight knots after rotation speed. Continue to hold back pressure until the desired angle of attack is approached, refer to figure 11-10 for a flap setting of 40° and figure 11-14 for a flap setting of 25°, then reduce back pressure to prevent overshooting the desired angle-of-attack. After the angle-of-attack is obtained, note attitude and horizon reference, then maintain this horizon reference as speed is increased to the approximate climb speed (at 50 feet and above) noted in figure 11-11 for 40 degree flaps, figure 11-15 for 25 degree flaps. The climb-out speed should be held until obstacle that is close-in is cleared. Speed should be increased to flap limit speed as rapidly as possible with a moderate rate of climb.

Data are based on maximum afterburner thrust with air conditioning on, engine oil cooler ejectors, nacelle vent ejectors, and hydraulic oil cooler ejectors open, when the aircraft weight is on the gear.

The rate of climb potential presented herein is based on free air (no ground effect).

TAKEOFF DISTANCES

Sample Problem

Enter figure 11-9 at the gross weight of 83,000 lbs, Ⓐ and follow the chase line horizontally left to the runway temperature line of 75°F (interpolated) Ⓑ, then project vertically upward to the pressure altitude line of 500 feet (interpolated) Ⓒ, then project right to the wind baseline Ⓓ and follow the headwind guide lines to 10 knots Ⓔ, then horizontally to the slope baseline Ⓕ following the uphill guide lines to 1% Ⓖ and read a ground run of 3,700 feet Ⓗ. To obtain a ground run distance corrected for center of gravity location, enter figure 11-10 at the ground run distance previously obtained (3,700 feet) Ⓐ for a clean aircraft with C.G. at 28% MAC. Project vertically to 27% MAC (interpolated) Ⓑ, then right to Ⓒ and read a corrected ground run distance of 3,700 feet.

TAKEOFF SPEED

Sample Problem

To obtain the takeoff speed for the given conditions, enter the top portion of figure 11-10 at 83,000 pounds gross weight Ⓐ, project vertically to intersect the 27% C.G. line (interpolated) Ⓑ then project horizontally to the left and read a takeoff speed of 148 KIAS Ⓒ. Rotation speed is 8 knots less than take-off speed or 140 KIAS.

CLIMB-OUT DISTANCE

The Climb-out Distance data presented in figures 11-11, 11-12, 11-15 and 11-16 are for flap settings of 40° and 25°, and are distances from unstick speed to an obstacle height in the vicinity of takeoff line of flight. To obtain the total takeoff distance for any given condition, the distance obtained from this chart must be added to the ground roli distance obtained from the appropriate charts.

Sample Problem

To obtain the climb-out distance required to clear a 150-foot obstacle, use figure 11-11, at the given gross weight of 83,000 pounds Ⓐ and project horizontally left to the runway temperature line of 75°F (interpolated) Ⓑ, then proceed vertically to the pressure altitude line of 500 feet (interpolated) Ⓒ, then project horizontally to the right to the obstacle base line Ⓓ. Move parallel to the guide lines to the obstacle height of 150 feet Ⓔ, then project horizontally to the right to the wind baseline Ⓕ and follow parallel to the headwind guide lines to 10 knots Ⓖ. Project to the right and read a distance from takeoff of 2300 feet Ⓗ.

CLIMB-OUT DISTANCE AND SPEED

Climb-out Distance and Speed data are presented in figures 11-12 and 11-16 for flap settings of 40° and 25°, respectively. The distance data in these charts are distance corrections for C.G. changes to data presented in figures 11-11 and 11-15. The climb-out speed data presented in figures 11-12 and 11-16 are the desired speeds at 50 feet above the runway, immediately after takeoff. Speed corrections for the effect of C.G. changes are indicated.

Sample Problem

To obtain a distance from takeoff corrected for center of gravity location, enter figure 11-12 at the distance of 2300 feet Ⓐ obtained for the clean aircraft with the C.G. at 28% MAC and project vertically upward to intersect the 27% MAC C.G. (interpolated) Ⓑ. Proceed horizontally to the right to intersect the line Ⓒ at 2,450 feet. The total distance to clear a 150-foot obstacle is the sum of the ground run and the climb-out distance or 4,000 feet + 2,400 feet = 6,400 feet.

The climb-out speed for the given conditions may be determined by guide lines in the upper portion of figure 11-12 to be 156 knots.

CRITICAL FIELD LENGTH

Critical Field Length data are presented in figures 11-17 and 11-19 for flap settings of 40° and 25°, respectively. The data are based on maximum afterburner thrust, and dry, hard-surface runway. The effects of runway surface winds and slope for critical field length are also presented.

Sample Problem (40° Flaps)

To determine critical field length for the given problem, enter the gross weight scale of figure 11-17 with a gross weight of 83,000 pounds Ⓐ, and proceed horizontally to a temperature of 75°F Ⓑ, and vertically to a pressure altitude of 500 feet Ⓒ, then horizontally to the baseline Ⓓ, and correct parallel to the slope for a headwind of 10 knots, then proceed to the baseline Ⓔ, and correct for an uphill slope of 1% Ⓕ, and read a critical field length of 5,900 feet for a C.G. of 28% MAC.

Sample Problem (25° Flaps)

To determine critical field length, enter the gross weight scale of figure 11-19 with a gross weight of 91,000 pounds and proceed as with the 40° flap example reading a value of 8400 feet at Ⓕ.

CRITICAL FIELD LENGTH - C.G. CORRECTION

Critical Field Length - C.G. Correction data are presented in figure 11-18 and 11-20. The data from figures 11-17 and 11-19 can be corrected for the effect of C.G. changes.

Sample Problem (40° Flaps)

To correct for C.G., enter figure 11-18 with a critical field length of 5,900 feet Ⓐ, and proceed vertically to a C.G. of 27% MAC Ⓑ, and horizontally to Ⓒ, and read a corrected field length of 6,000 feet.

Find:

Using a 40° Flap Setting at a gross weight of 91,000 lbs.

Ground Run

Total Distance over 150 ft. obstacle

Rotation speed

Takeoff speed

Climb-out speed

Critical Field Length (dry)

Critical Engine failure speed (dry)

Refusal Speed (dry)

Continuation Speed

G-no-Go speed

Acceleration Check Speed

Sample Problem (25° Flaps)

To correct for C.G., enter figure 11-20 with a critical field length of 8,400 feet (A), and proceed vertically to a C.G. of 27% MAC (B), then horizontally to (C), and read a corrected critical field length of 8,600 feet.

CRITICAL ENGINE FAILURE AND REFUSAL SPEED

The Critical Engine Failure and Refusal Speed chart, figure 11-21 is applicable for flap settings of 40° and 25° during takeoff. The data are based on maximum afterburner thrust and present information for an operational envelope of gross weights, altitudes and field lengths. Also provided are corrections for the effect of runway surface winds.

Sample Problem

Enter figure 11-21 with a gross weight 91,000 pounds (A), and project horizontally to a temperature of 75°F (B), and vertically to a pressure altitude of 500 feet (C) and, horizontally to a runway length (critical field length) of 8,600 feet (D), then to the wind baseline (E), and parallel to the headwind guide lines to 10 knots and read a critical engine failure speed of 157 KIAS (F). For a runway length of 12,000 feet (field length) (H), the refusal speed is 178 KIAS.

VELOCITY DURING TAKEOFF GROUND RUN

The Velocity During Takeoff Ground Run of figures 11-22 and 11-25 presents lines of accelerations for two and single-engine maximum thrust operation, respectively. The charts can be used to determine distances required to accelerate from one speed to another along an acceleration line. In any case, a starting point must first be established prior to use of chart. As an example, after a takeoff distance and speed are determined, these values can be used to establish a point on the chart. Then any change along an acceleration line from this point will result in a speed and distance change. As an aid to establish starting points for calculations on figure 11-25, figures 11-23 and 11-24 are presented.

Sample Problem

The go-no-go and acceleration check speeds are determined from figure 11-22 by entering with the takeoff distance of 4,800 feet (A), and takeoff speed 165 KIAS (B) or 168 knots true ground speed. Follow the chase around lines to the intersection point and follow parallel to the guide lines to the refusal speed of 178 KIAS (C) or 182 knots true ground speed and obtain a refusal distance of (D) of 5,800 feet. Since the first runway marker before this 5,800 feet is at 5,000 feet (E), the go-no-go speed (F) is 168 KIAS or 171 knots true ground speed. Following the guide lines from the 5,000-foot marker down to the 3,000-foot runway marker (G) before go-no-go, the acceleration check speed of 134 KIAS (H) or 137 knots true ground speed is read. By definition, the acceleration check speed is made at a point 2,000 feet less than the go-no-go distance marker. The true ground speed values are used to determine distance and time when desired.

RUNWAY STOPPING DISTANCE.

Stopping distance data are presented in figures 11-25A (sheets 1 and 2) and 11-25B (sheets 1 and 2). Stopping distance from an abort velocity can be determined for either wet or dry runways with or without barriers. Brake energy limits, as well as braking or coasting are indicated on the charts. The coasting portion of the charts are based on all wheels in contact with runway with elevator trimmed for take-off. Some improvement in coasting deceleration could be obtained by maximum back pressure on control stick but not lifting nose wheel off runway. No braking pressures are applied during coasting deceleration until brake limits are intersected. Chase around lines are presented to indicate the use of charts.

Sample Problem

Given:

Gross weight = 91,000 pounds
Temperature = 75° F
Runway Altitude = 500 feet (pressure altitude)
Headwind = 10 knots
Dry Runway without Barrier
Flaps = 25 degrees
Wing sweep = 16 degrees
Refusal speed = 178 knots

Find:

Stopping distance when using brake energy limit of 37.5 MILLION FT/LB.

TAKE-OFF/ABORT CRITERIA

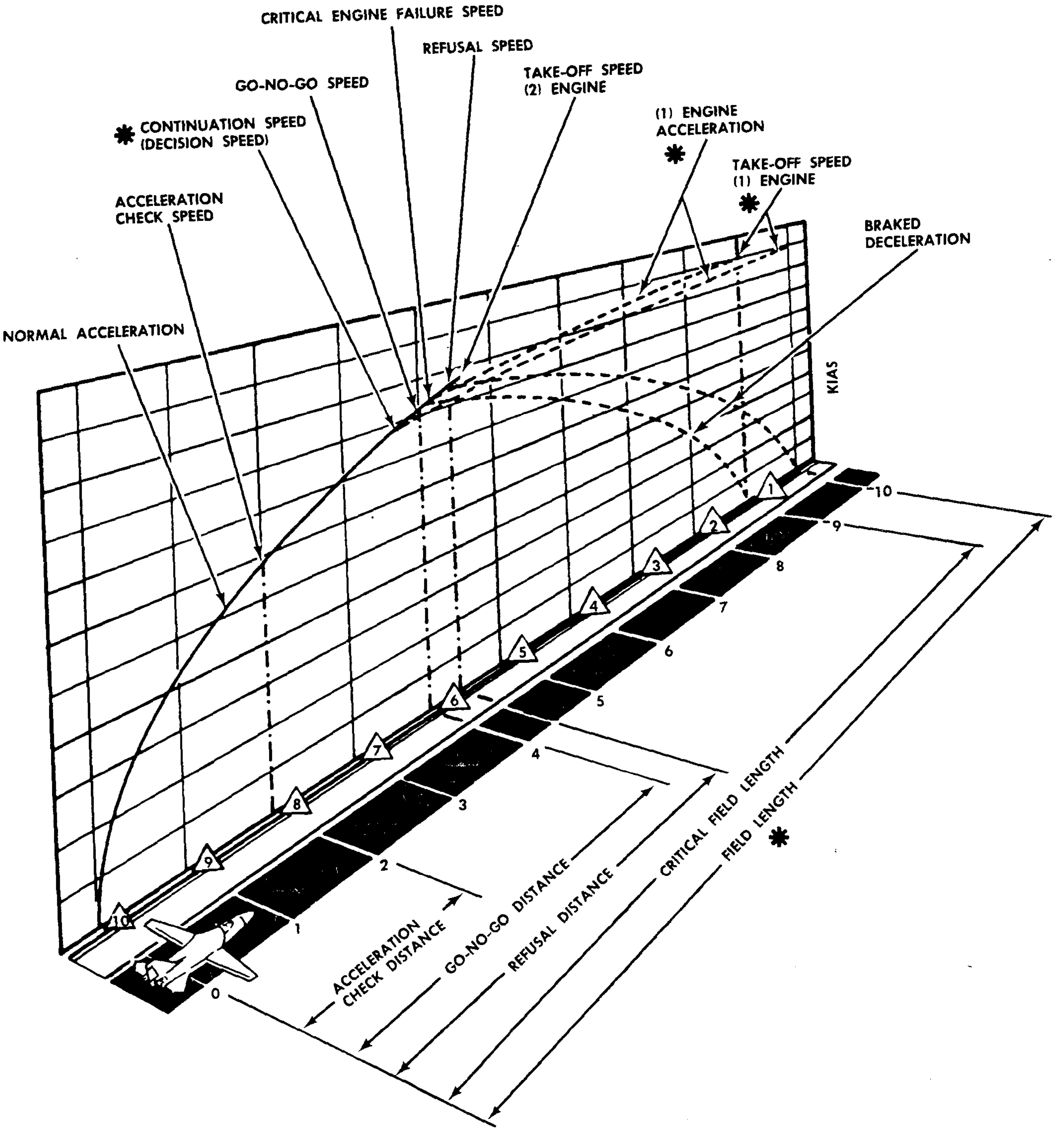

Figure 11-8

Solution:

Enter figure 11-25A (sheet 1) at given temperature of 75° F (A) and follow chase around through to 500 feet altitude (B), refusal speed of 178 KIAS - 10 knots headwind or 168 KIAS (C), gross weight of 91,000 pounds (D) and obtain stopping distance of 5390 feet (E). Observe that point (D) intersection is to right of 37.5 million foot-pound brake energy limit which indicates that to observe this limit reduce speed from 172 KIAS (D) to 147 KIAS (D') ground speed prior to brake application. Distance traveled during this coasting from point (D) to (D') is 5390-3000 feet or 2390 feet. Remaining stopping distance from brake application speed point of 147 ground speed (D') is 3000 feet (E'). It should be understood that after temperature and altitude corrections are applied to indicated airspeeds (C) and (C') the resultant speeds at (D) and (D') respectively are ground speeds.

TAKE-OFF DISTANCE

REMARKS
ENGINES: TF 30-P-12
WING SWEEP — 16°
40° FLAPS
MAXIMUM THRUST
DRY, HARD SURFACED RUNWAY
C.G. 28% MAC

DATE: 15 MARCH 1968
DATE BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

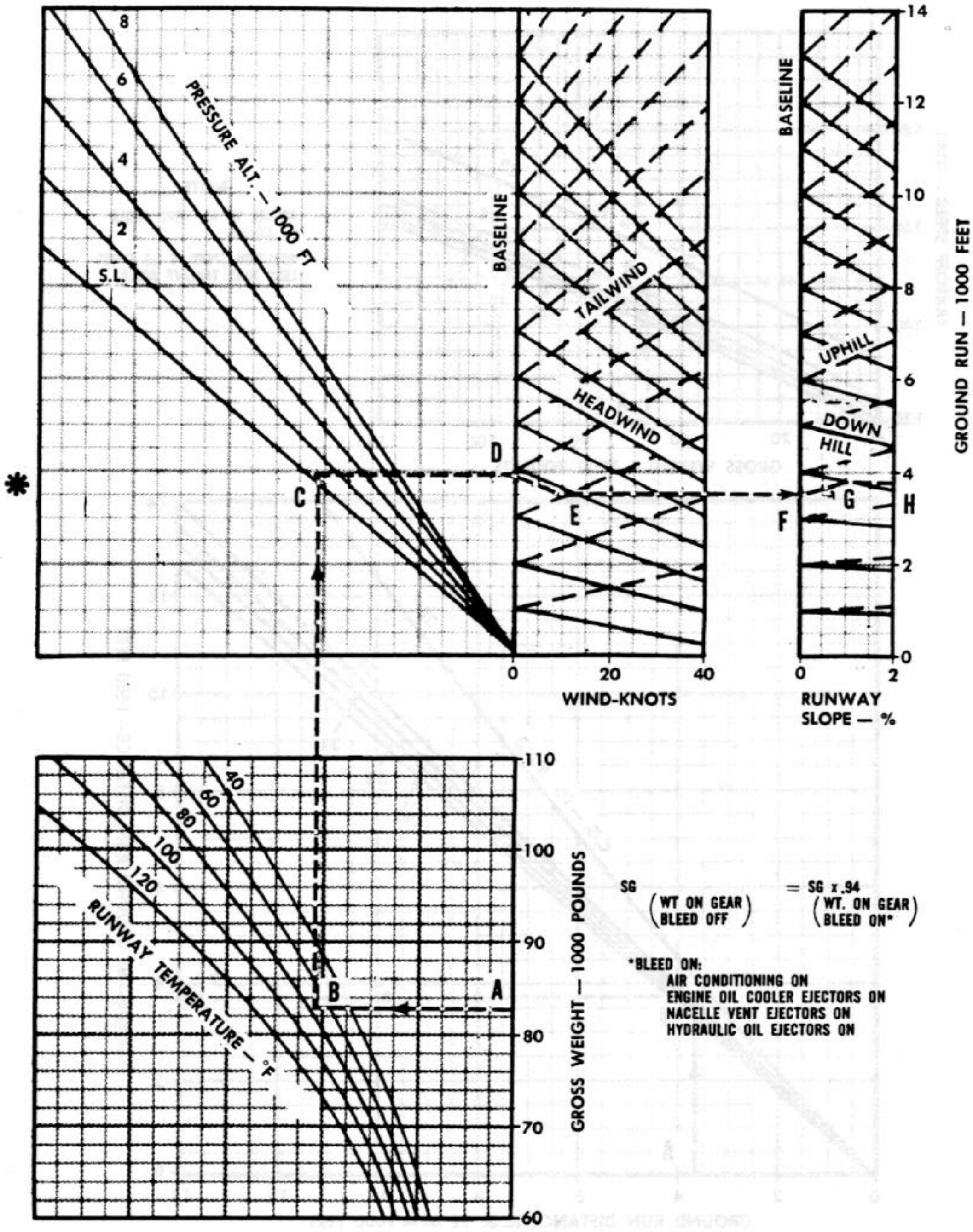

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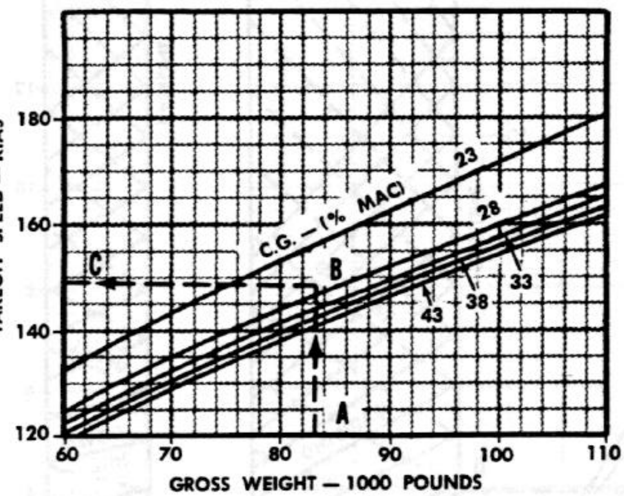
Figure 11-9

TAKE-OFF DISTANCE AND SPEED— C. G. CORRECTION

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINES: TF 30-P-12
WING SWEEP = 16°
40° FLAPS
TWO ENGINES OPERATING
MAXIMUM THRUST
DRY, HARD SURFACED RUNWAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



NOTE

ROTATE TO 13° WING ANGLE OF ATTACK

ROTATION SPEED IS 8.0 KNOTS LESS THAN TAKEOFF SPEED

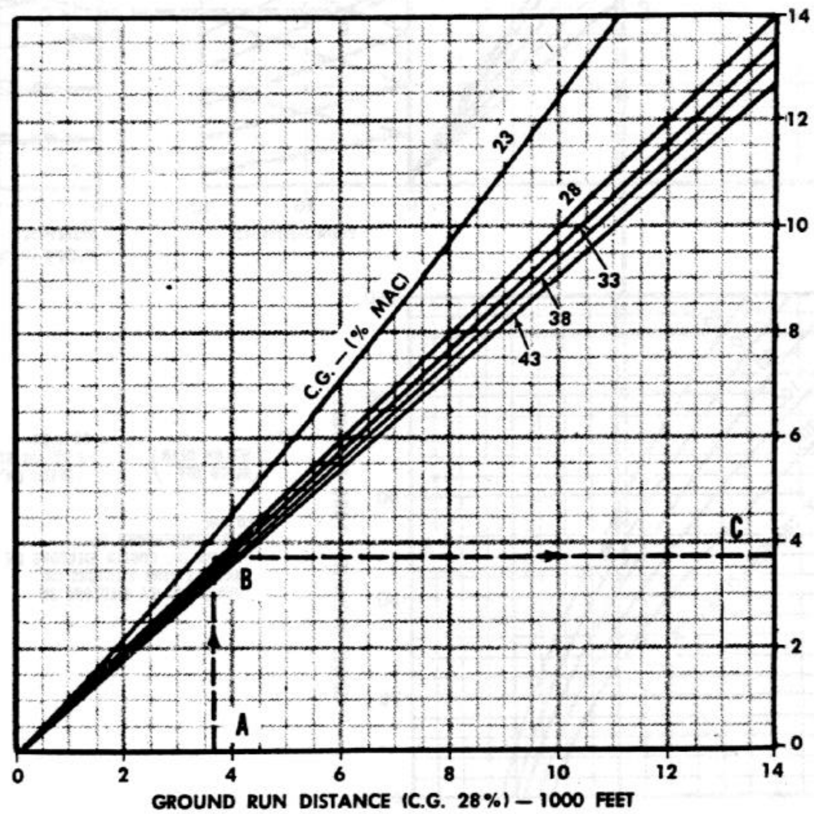


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Figure 11-10

CLIMBOUT DISTANCE

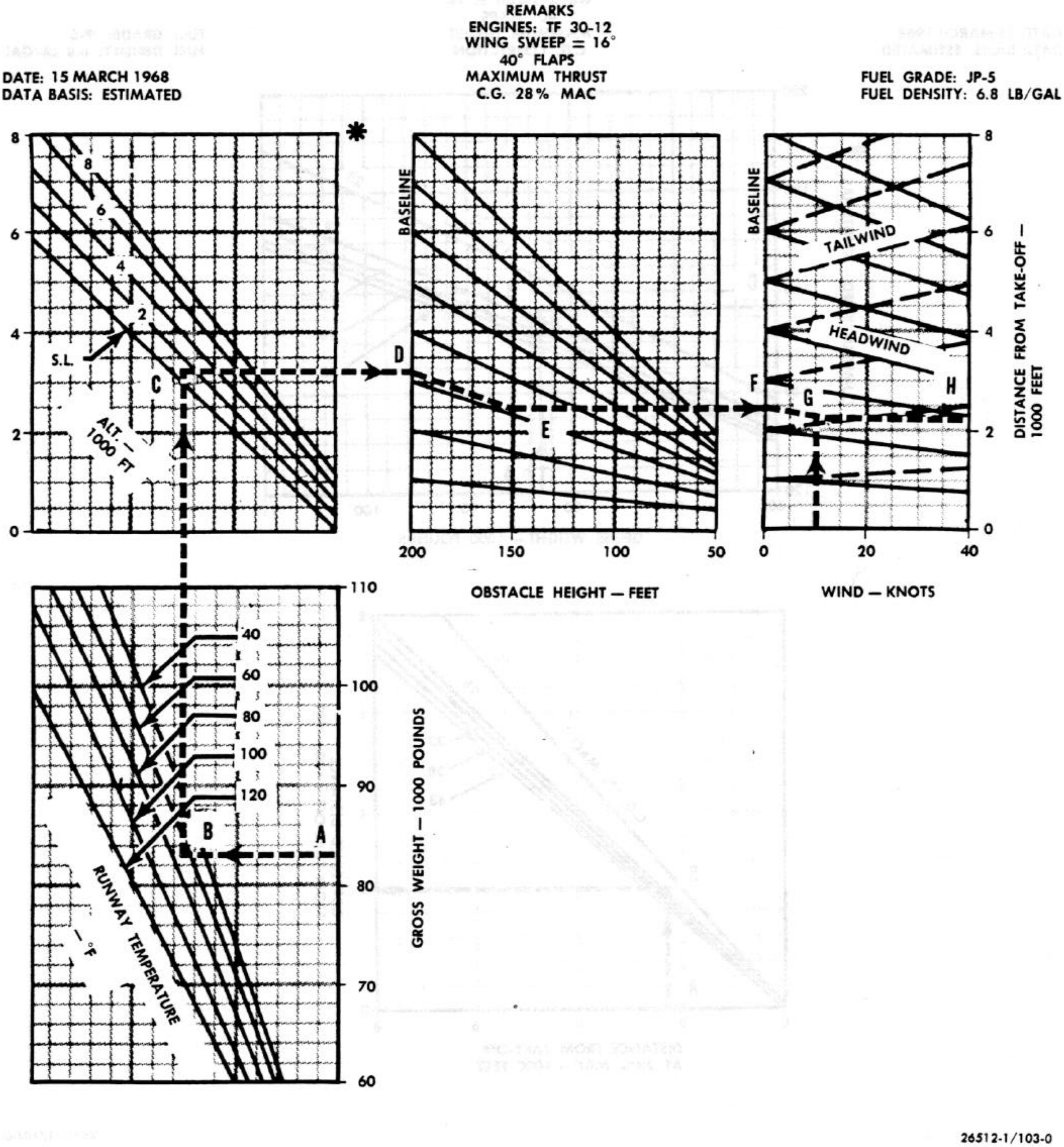


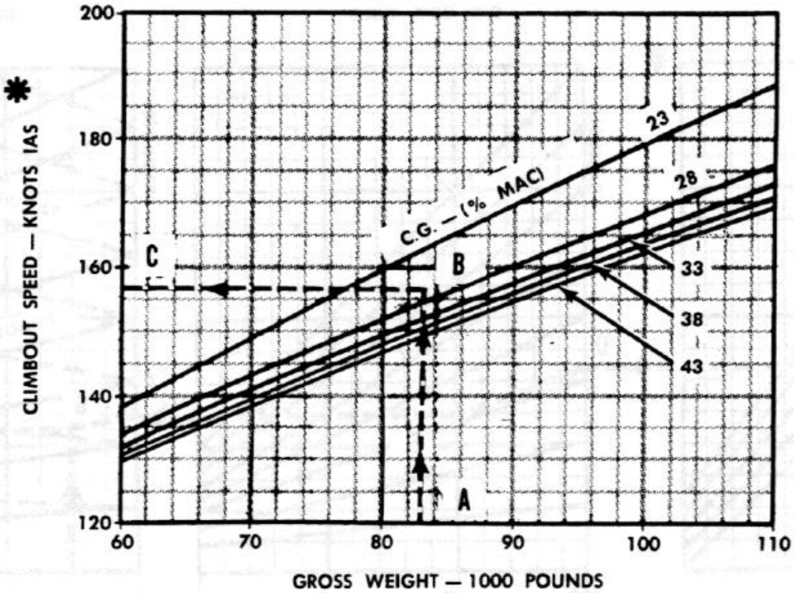
Figure 11-11

CLIMBOUT DISTANCE AND SPEED

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINES: TF 30-12
WING SWEEP = 16°
40° FLAPS
MAXIMUM THRUST
C.G. CORRECTION

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



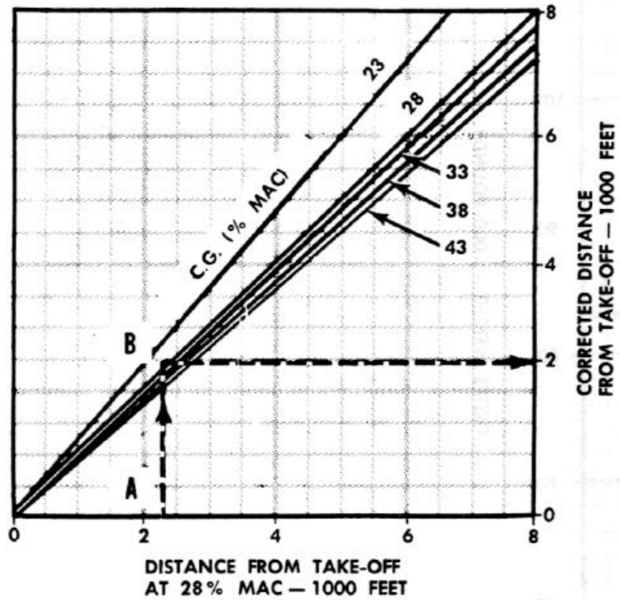


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Figure 11-12

TAKE-OFF DISTANCE

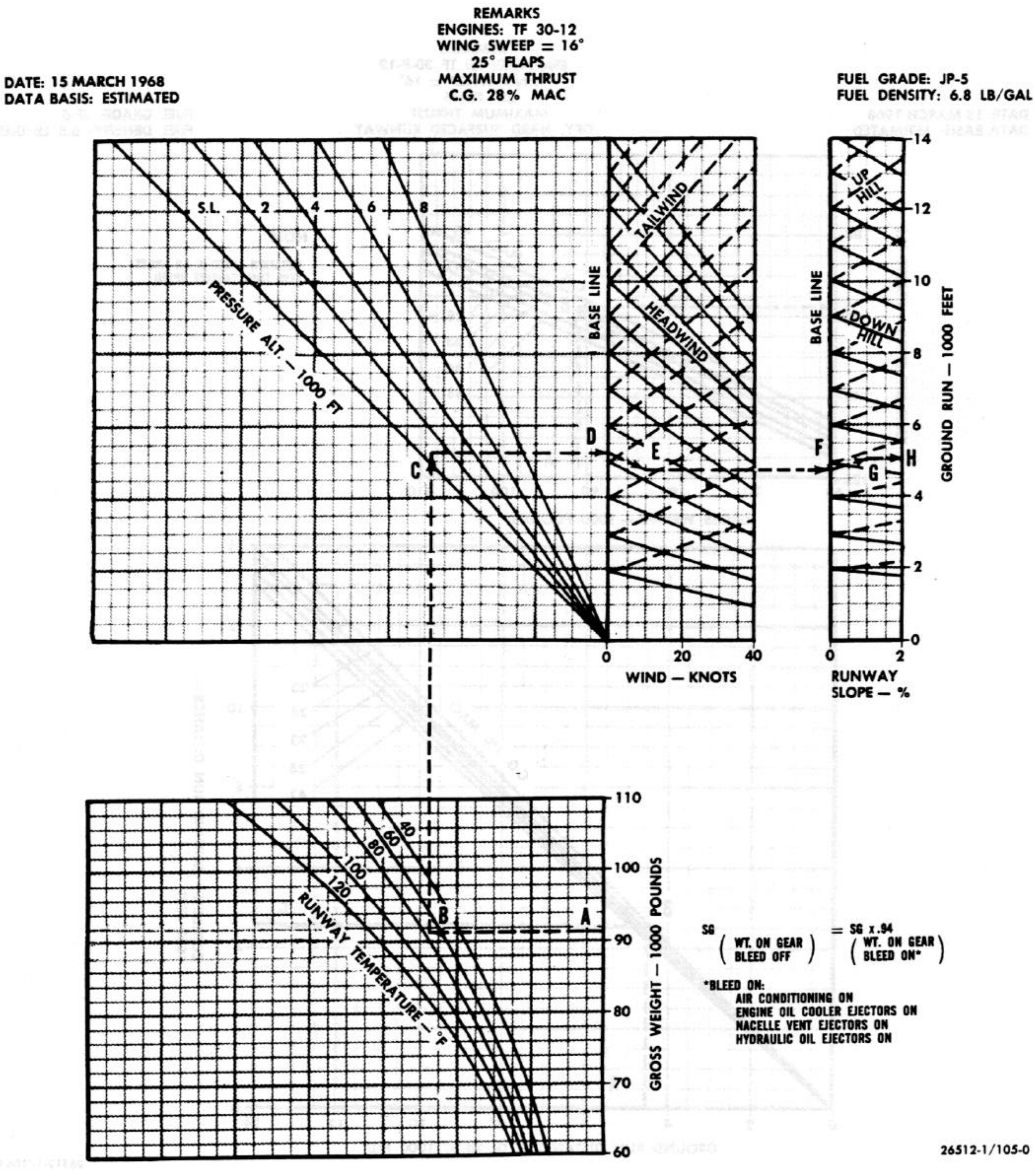


Figure 11-13

TAKE-OFF DISTANCE AND SPEED— C. G. CORRECTION

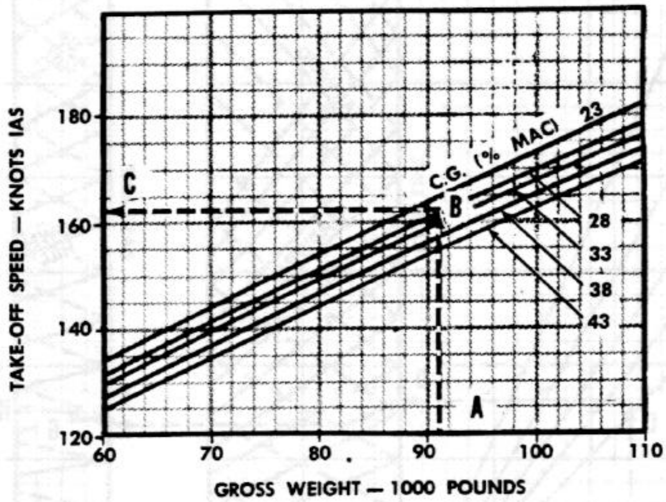
DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF 30-P-12
WING SWEEP = 16°
25° FLAPS
MAXIMUM THRUST
DRY, HARD SURFACED RUNWAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

NOTE

ROTATION SPEED IS 8.0 KNOTS LESS THAN TAKEOFF SPEED



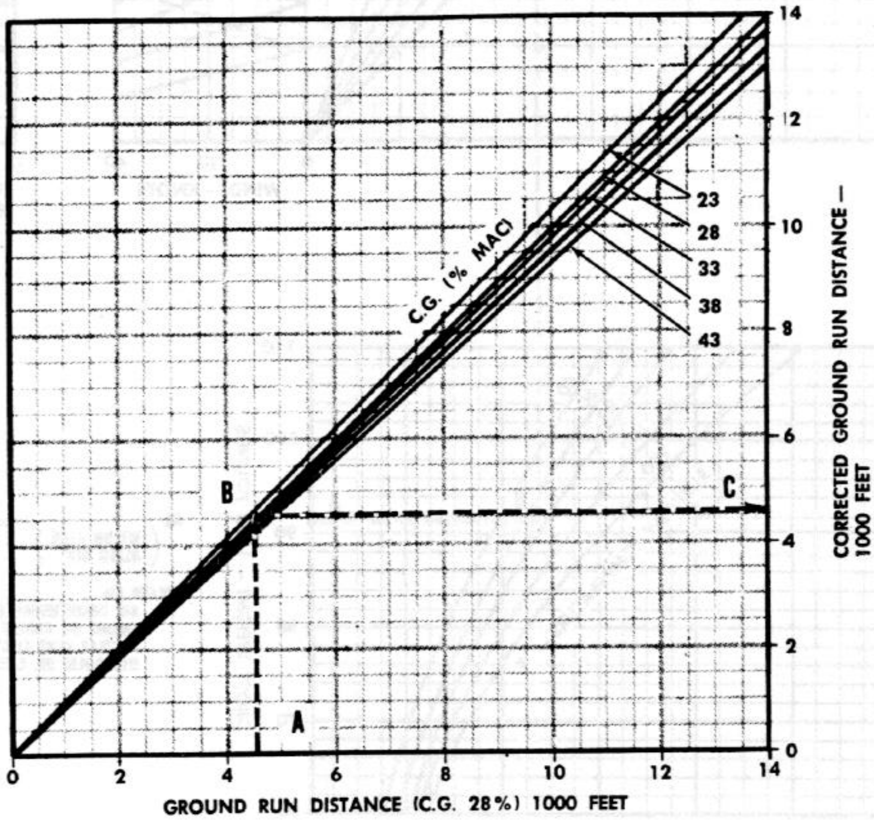


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Figure 11-14

CLIMBOUT DISTANCE

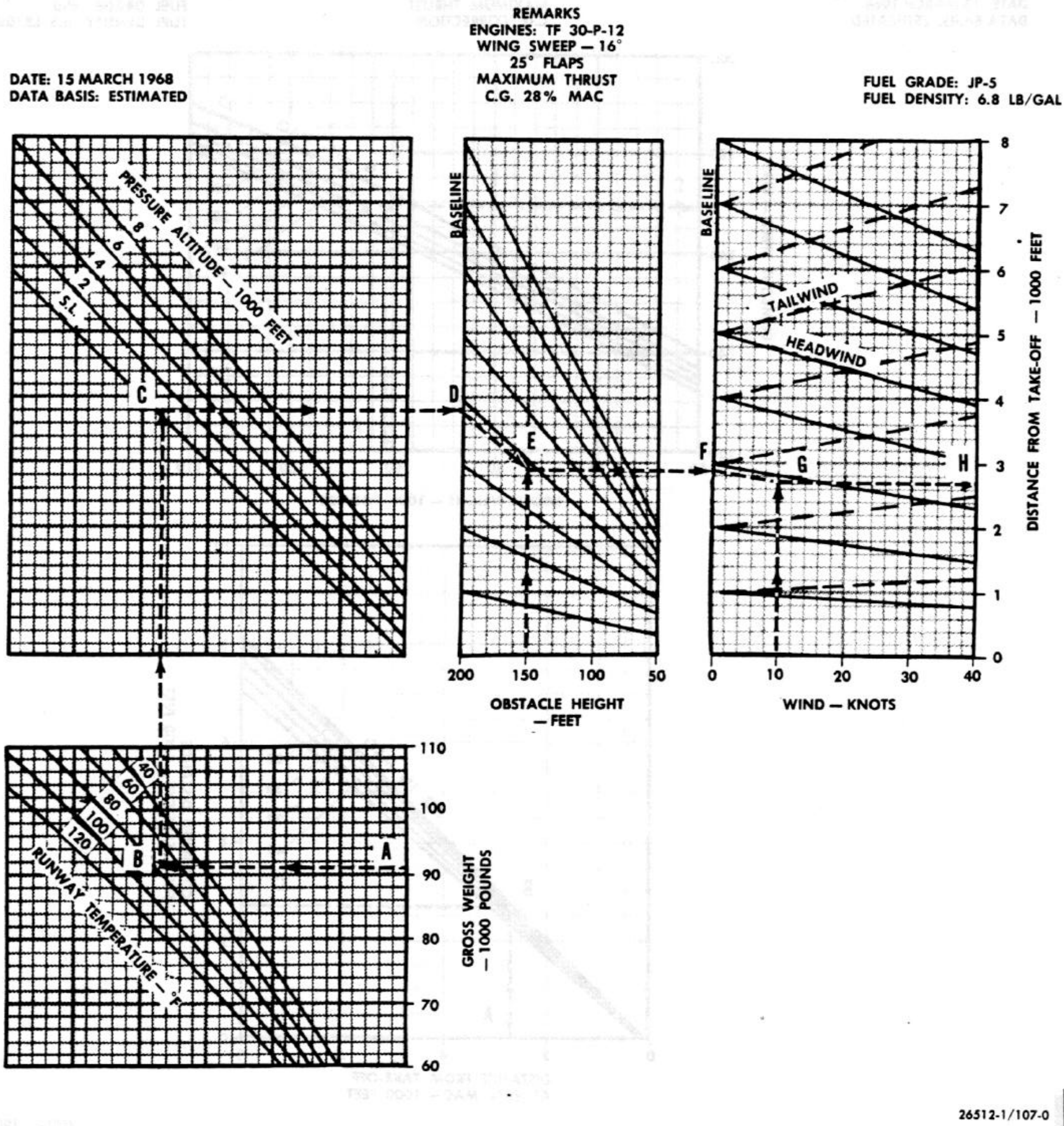


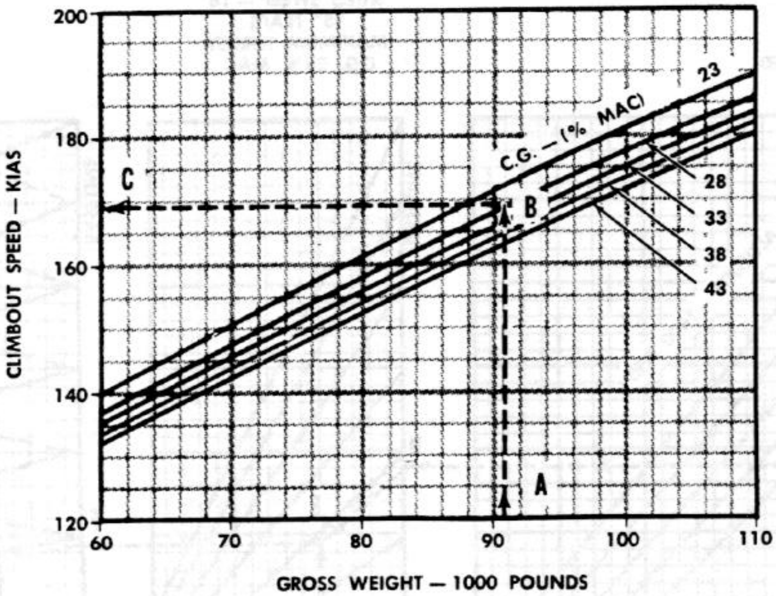
Figure 11-15

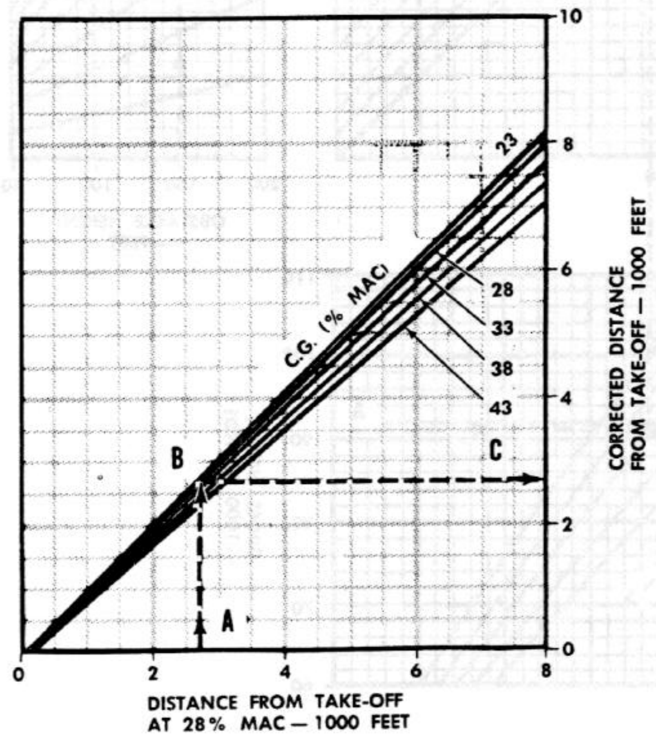
CLIMBOUT DISTANCE AND SPEED

REMARKS
ENGINE(S): TF30-P-12
WING SWEEP = 16°
25° FLAPS
MAXIMUM THRUST
C.G. CORRECTION

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL





26512-1/108-0

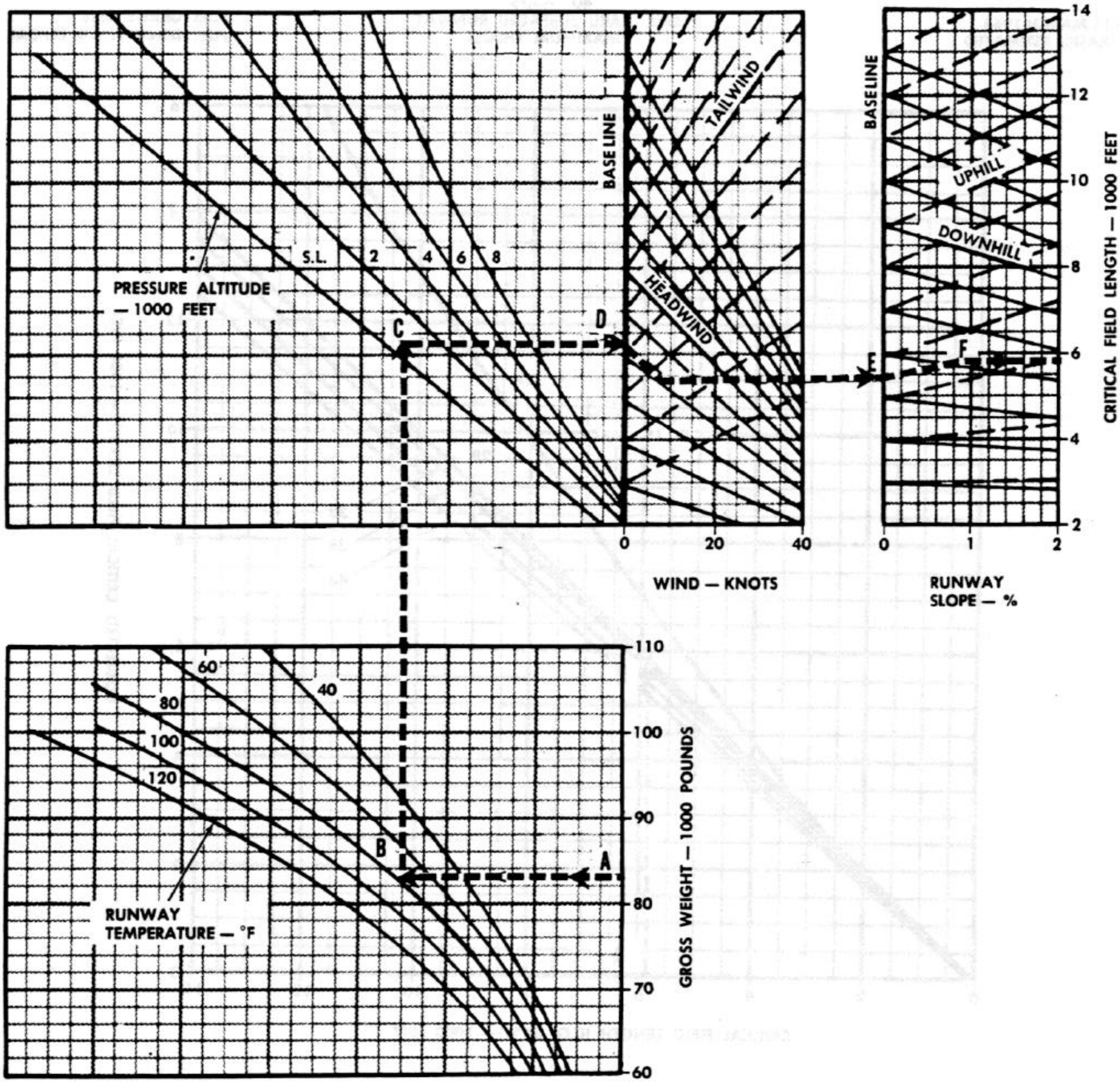
Figure 11-16

CRITICAL FIELD LENGTH

REMARKS
ENGINES: TF 30-P-12
WING SWEEP = 16°
40° FLAPS
MAXIMUM THRUST
DRY, HARD SURFACED RUNWAY
C.G. 28% MAC

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/109-0

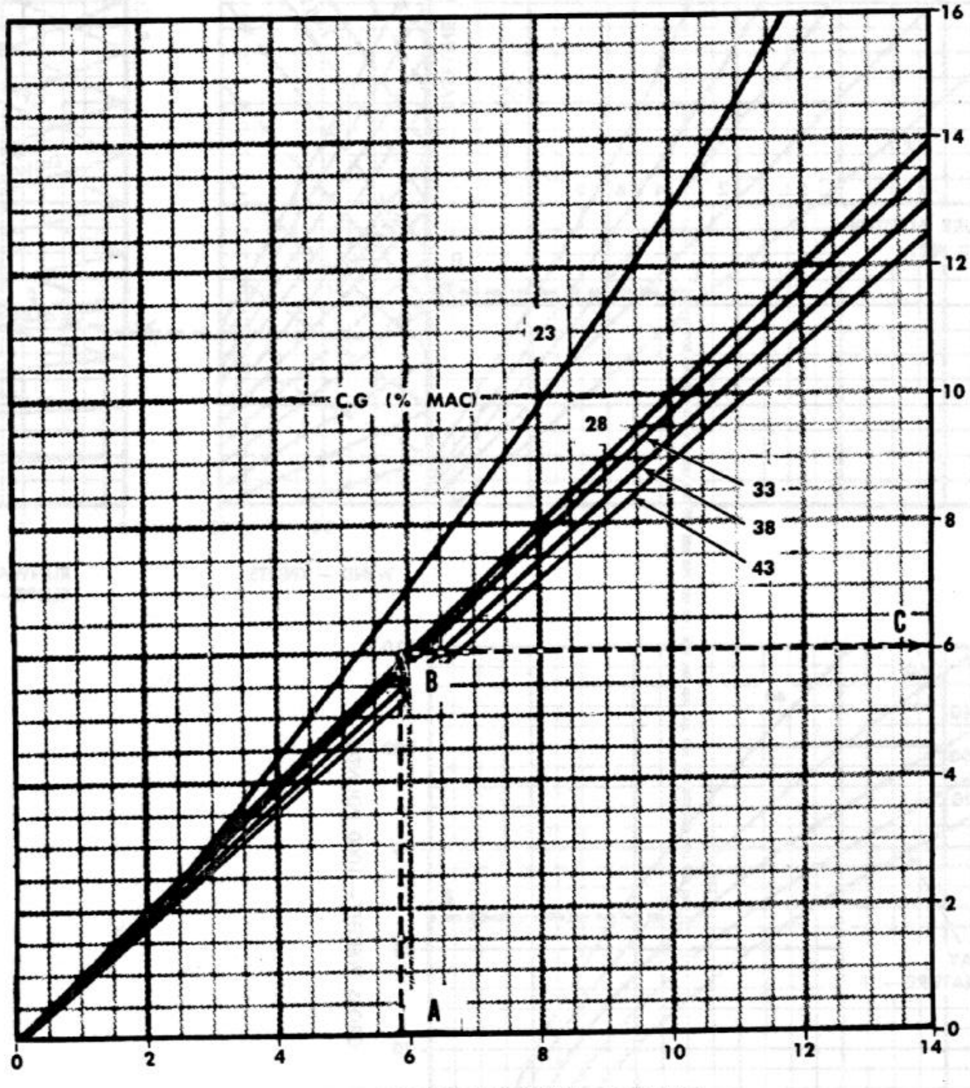
Figure 11-17

CRITICAL FIELD LENGTH— C. G. CORRECTION

REMARKS
ENGINE(S): (2) TF30-P-12
WING SWEEP = 16°
40° FLAPS
DRY, HARD SURFACED RUNWAY
MAXIMUM THRUST

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/110-0

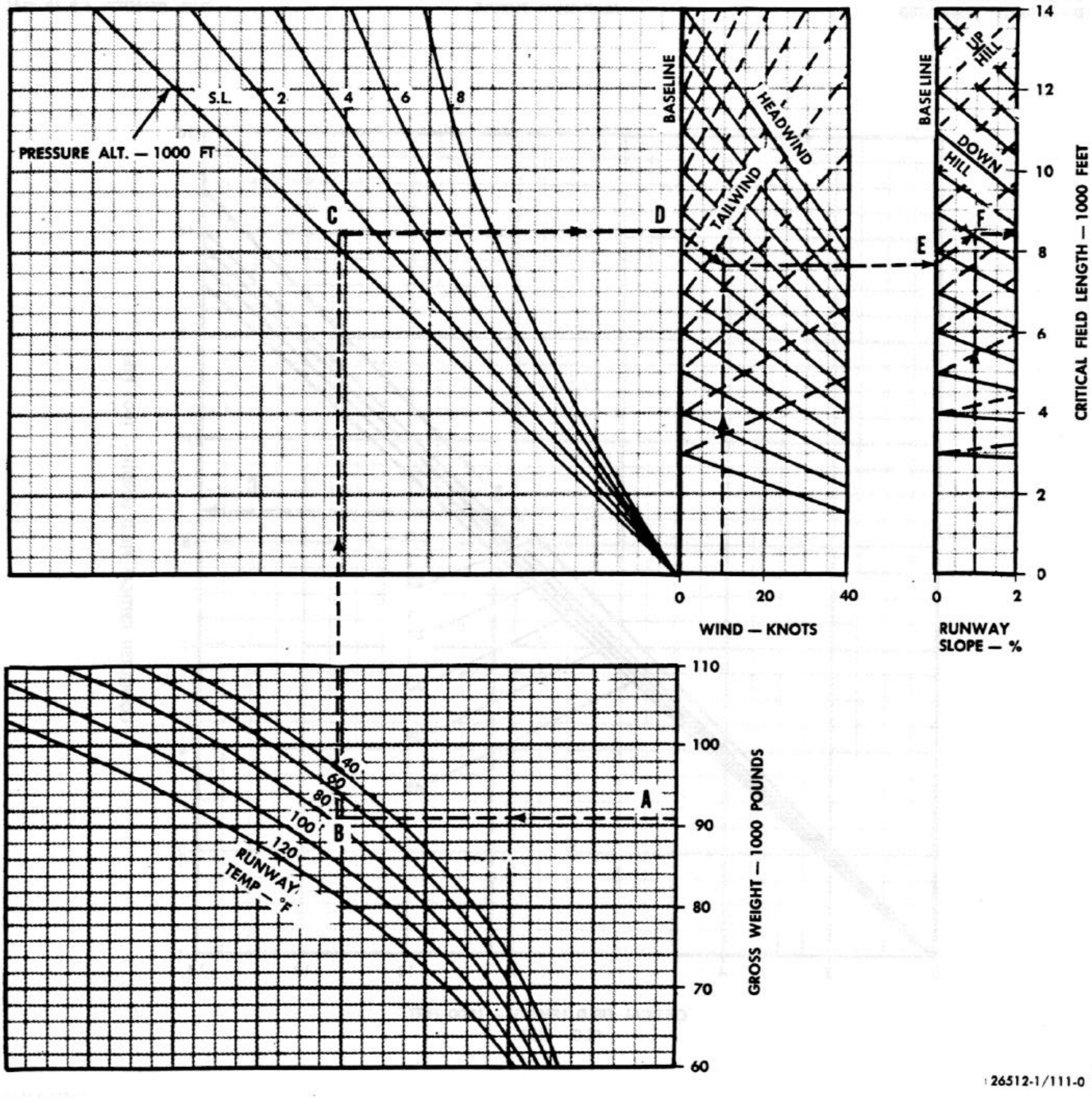
Figure 11-18

CRITICAL FIELD LENGTH

REMARKS
ENGINES: TF 30-P-12
WING SWEEP = 16°
25° FLAPS
MAXIMUM THRUST
DRY, HARD SURFACED RUNWAY
C.G. 28% MAC

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/111-0

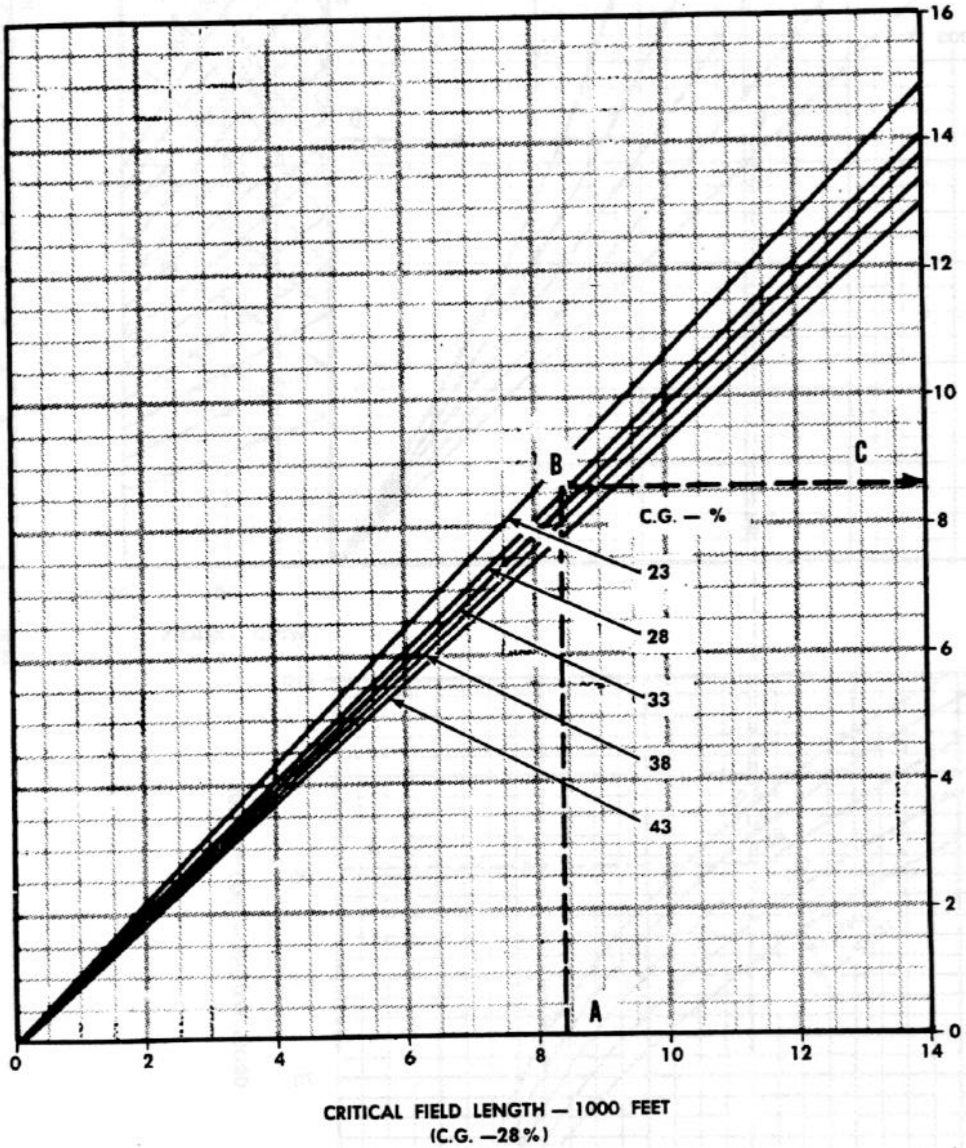
Figure 11-19

CRITICAL FIELD LENGTH— C. G. CORRECTION

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINES: TF 30 — P-12
WING SWEEP = 16°
25° FLAPS
MAXIMUM THRUST

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/112-0

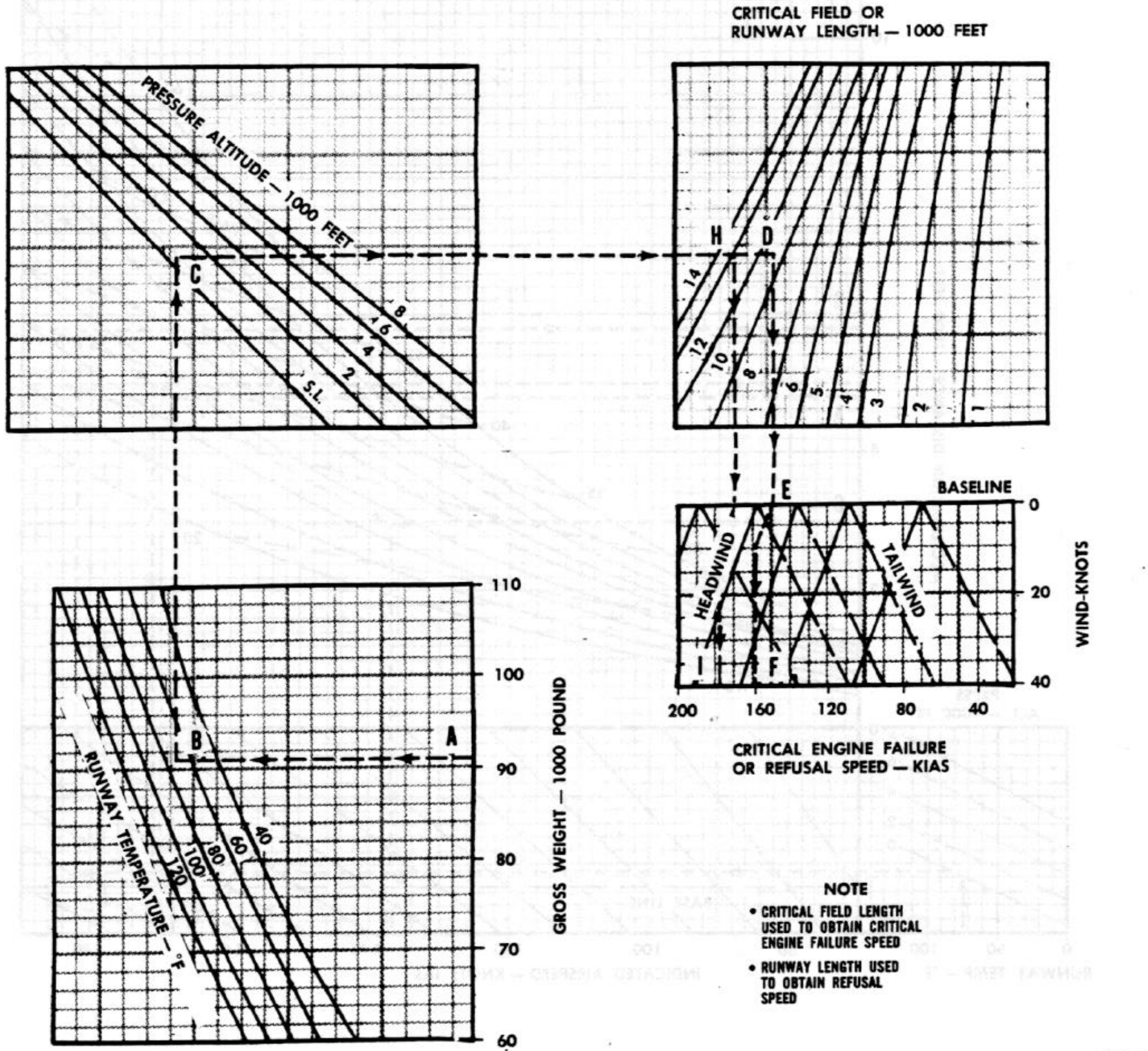
Figure 11-20

CRITICAL ENGINE FAILURE AND REFUSAL SPEEDS

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINES: TF 30-P-12
WING SWEEP — 16°
25°-40° FLAPS
MAXIMUM THRUST
DRY, HARD SURFACED RUNWAY
C.G. 28% MAC

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



NOTE
- CRITICAL FIELD LENGTH USED TO OBTAIN CRITICAL ENGINE FAILURE SPEED
- RUNWAY LENGTH USED TO OBTAIN REFUSAL SPEED

26512-1/113-0

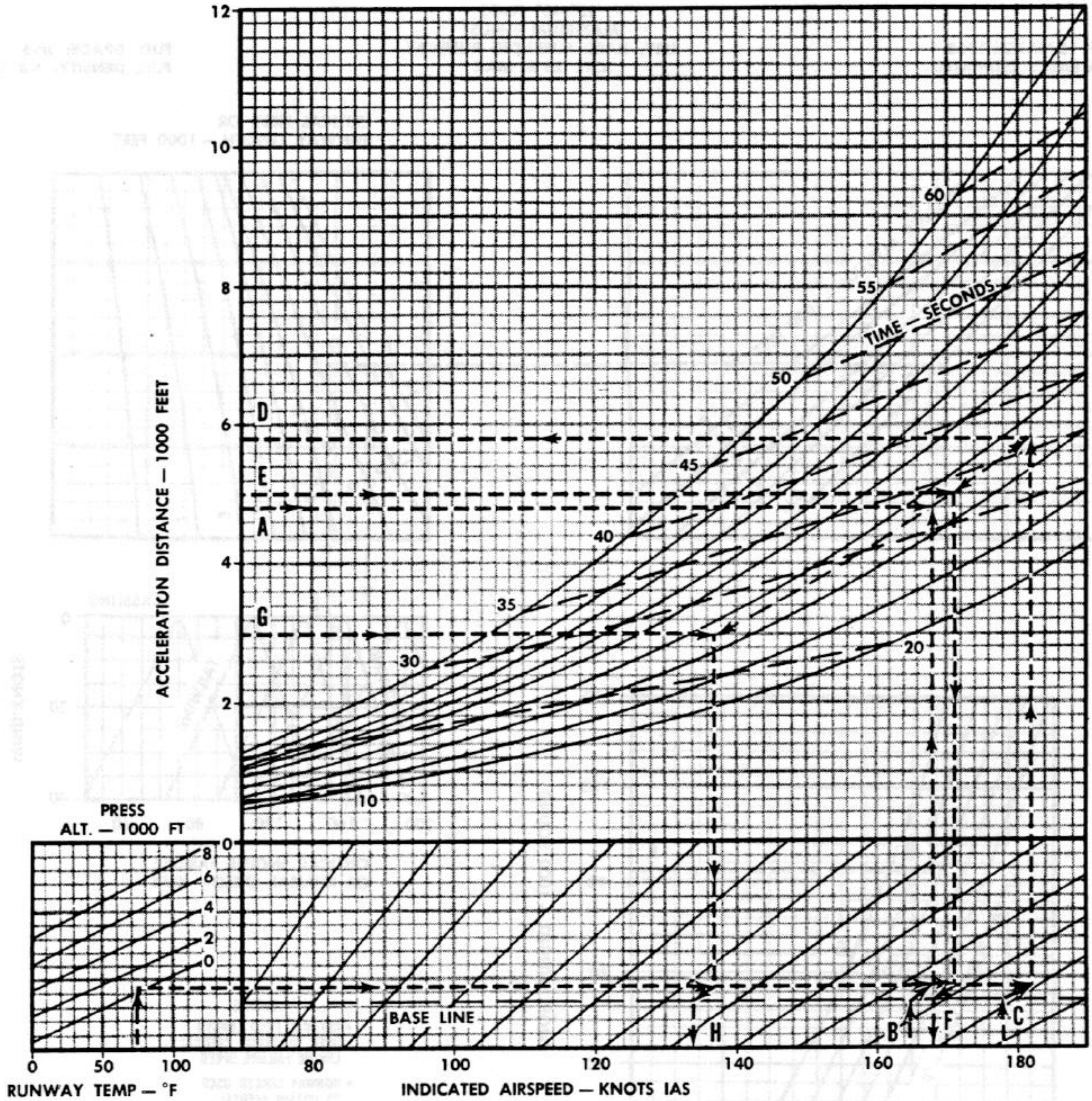
Figure 11-21

VELOCITY DURING TAKE-OFF GROUND RUN

REMARKS
ENGINE(S): (2) TF 30-P-12
WING SWEEP = 16°
25°-40° FLAPS
MAXIMUM THRUST
TWO ENGINES OPERATING
DRY, HARD SURFACED RUNWAY

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/114-0

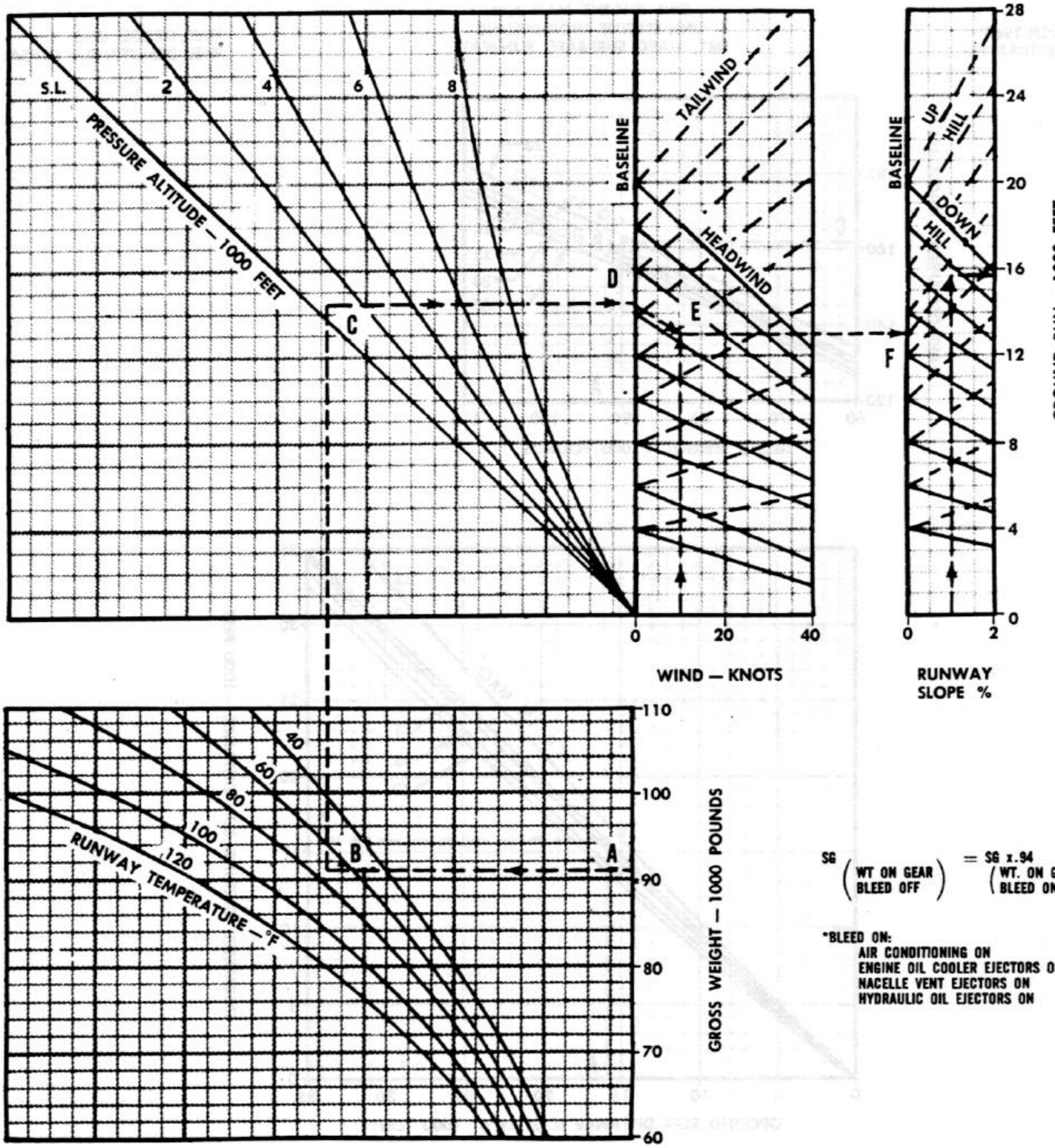
Figure 11-22

TAKE-OFF DISTANCE—(single engine)

REMARKS
ENGINES: TF 30-P-12
WING SWEEP = 16°
25° FLAPS
ONE ENGINE MAX A/B
ONE ENGINE WINDMILLING
C6 28% MAC

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



SG (WT ON GEAR BLEED OFF) = SG x.94 (WT. ON GEAR BLEED ON*)

*BLEED ON:
AIR CONDITIONING ON
ENGINE OIL COOLER EJECTORS ON
NACELLE VENT EJECTORS ON
HYDRAULIC OIL EJECTORS ON

26512-1/115-0

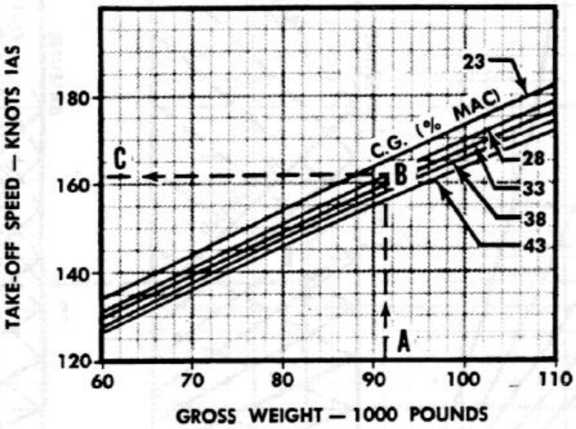
Figure 11-23

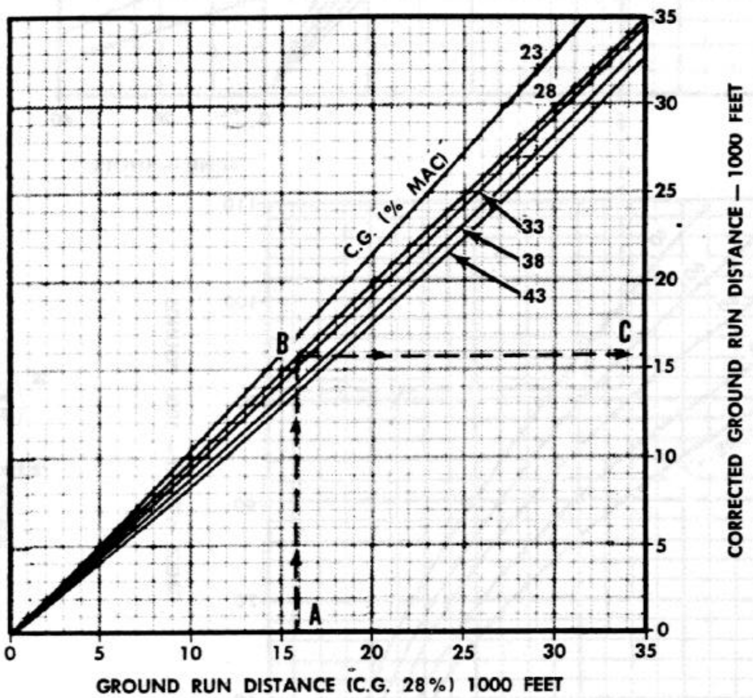
TAKE-OFF DISTANCE AND SPEED— C. G. CORRECTION (single engine)

REMARKS
ENGINES: TF 30-P-12
WING SWEEP = 16°
25° FLAPS
ONE ENGINE MAX A/B
ONE ENGINE WINDMILLING
DRY, HARD SURFACED RUNWAY

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL





26512-1/116-0

Figure 11-24

VELOCITY DURING TAKE-OFF GROUND RUN (single engine)

REMARKS
ENGINE(S): (2) TF 30-P-12
WING SWEEP = 16°
25° FLAPS
MAXIMUM THRUST
DRY, HARD SURFACED RUNWAY

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

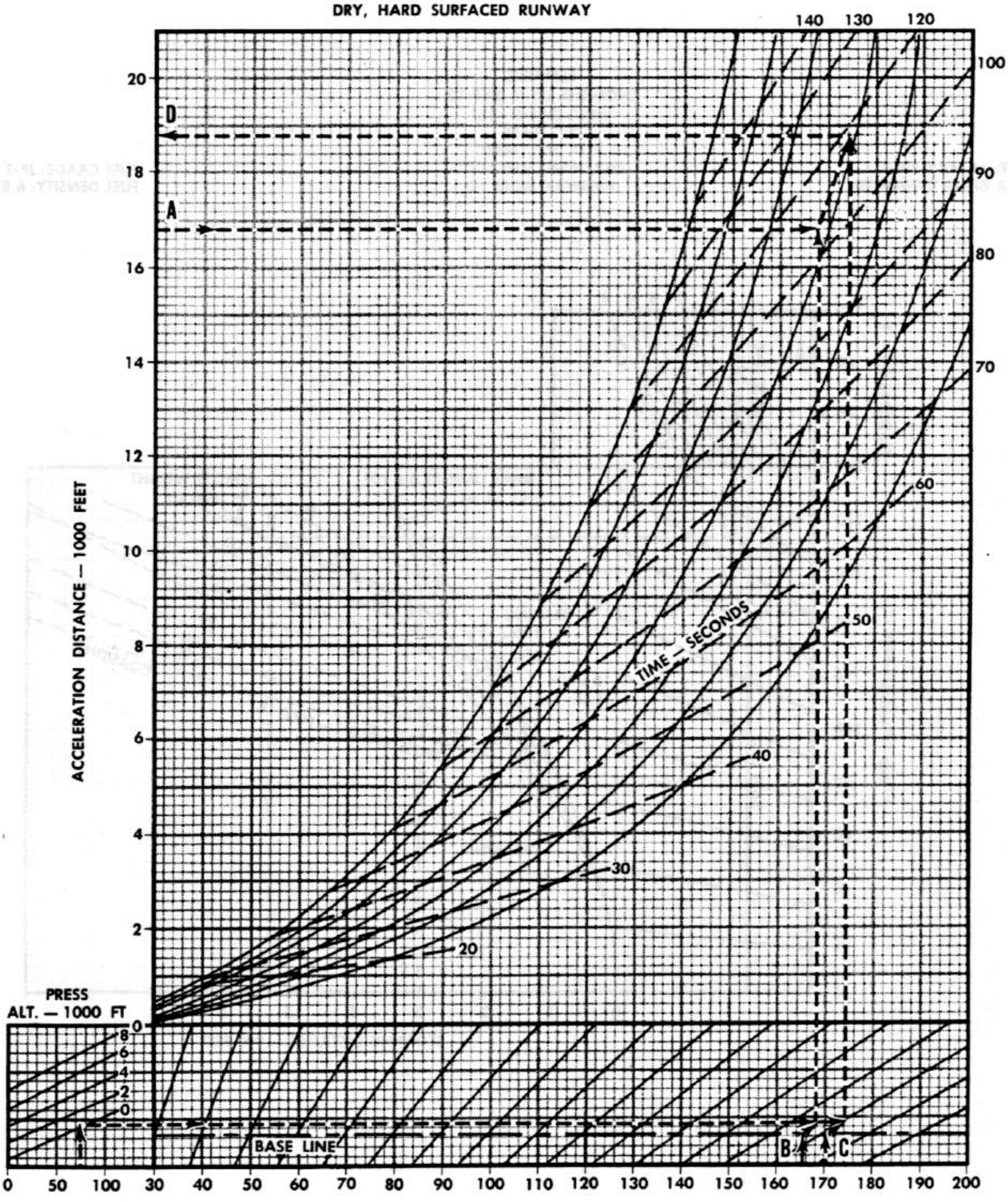


Figure 11-25

RUNWAY STOPPING DISTANCE

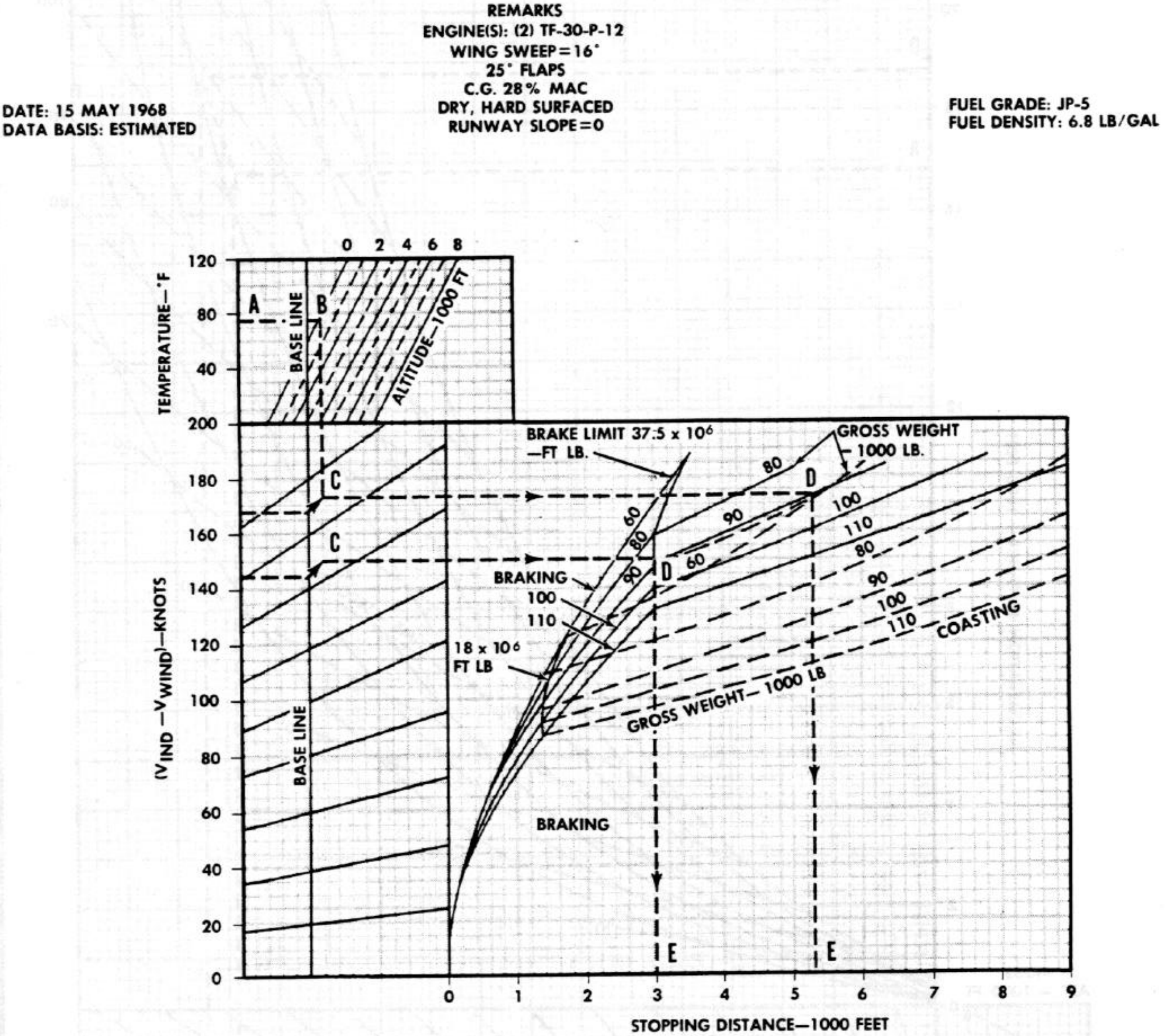


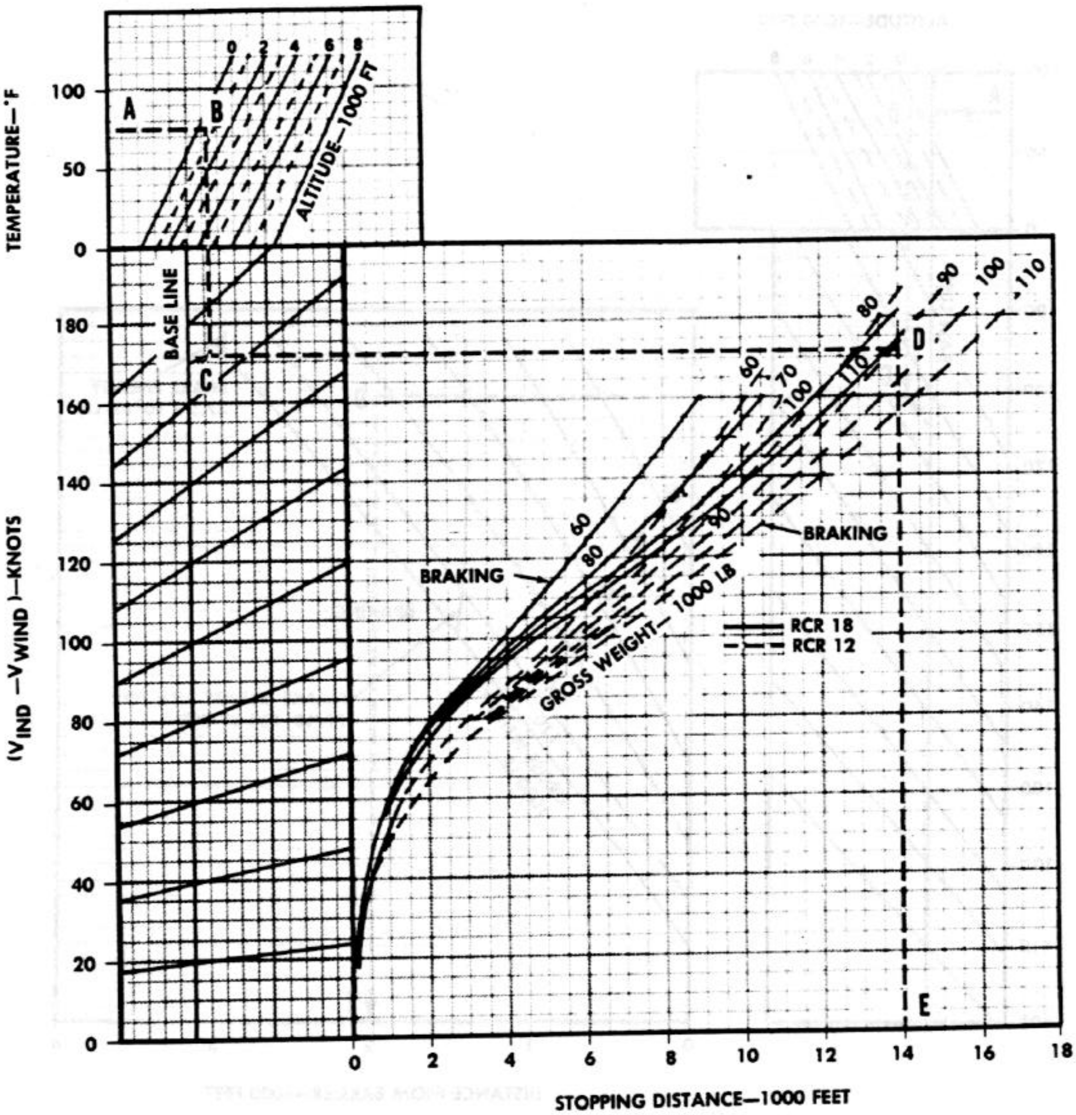
Figure 11-25A (sheet 1)

RUNWAY STOPPING DISTANCE

REMARKS
ENGINE(S): (2) TF-30-P-12
WING SWEEP=16°
25° FLAPS
C.G. 28% MAC
WET, HARD SURFACED
RUNWAY SLOPE=0

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/128.2-0

Figure 11-25A (sheet 2)

RUNWAY STOPPING DISTANCE (based on maximum engaging speed BAK 12 limit—1" cable)

REMARKS
ENGINES: TF-30-P-12
WING SWEEP=16°
25° FLAPS
C.G. 28% MAC
DRY, HARD SURFACED RUNWAY
RUNWAY SLOPE=0

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP—5
FUEL DENSITY: 6.8 LB/GAL.

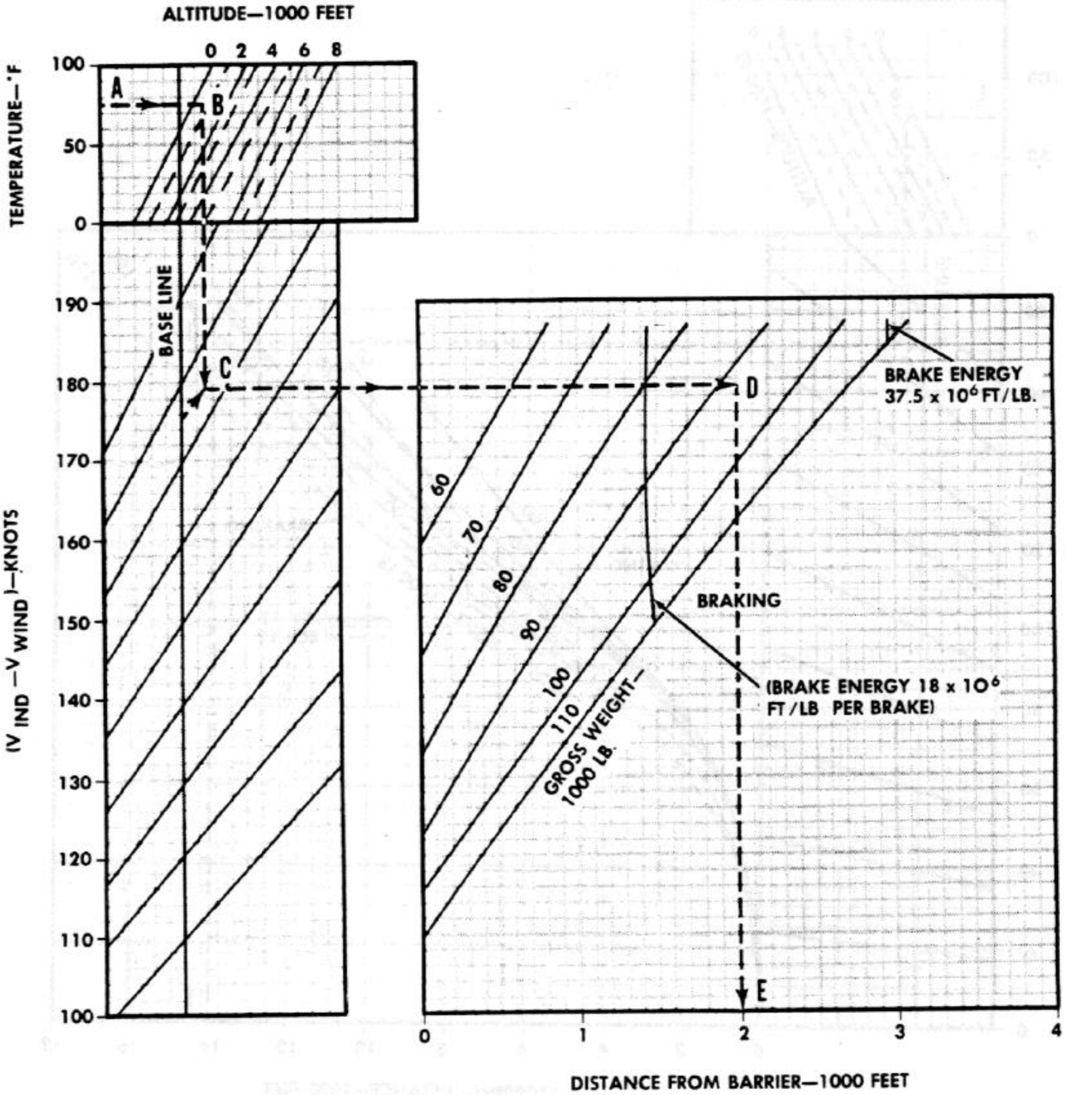


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Figure 11-25B (sheet 1)

RUNWAY STOPPING DISTANCE (based on maximum engaging speed BAK 12 limit— 1" cable)

REMARKS
ENGINES: TF-30-P-12
WING SWEEP=16°
25° FLAPS
C.G. 28% MAC
RUNWAY SLOPE=0
WET, HARD SURFACED RUNWAY

DATE: 15 MAY 1968
DATA BASIS: ESTIMATED

FUEL GRADE: JP—5
FUEL DENSITY: 6.8 LB/GAL.

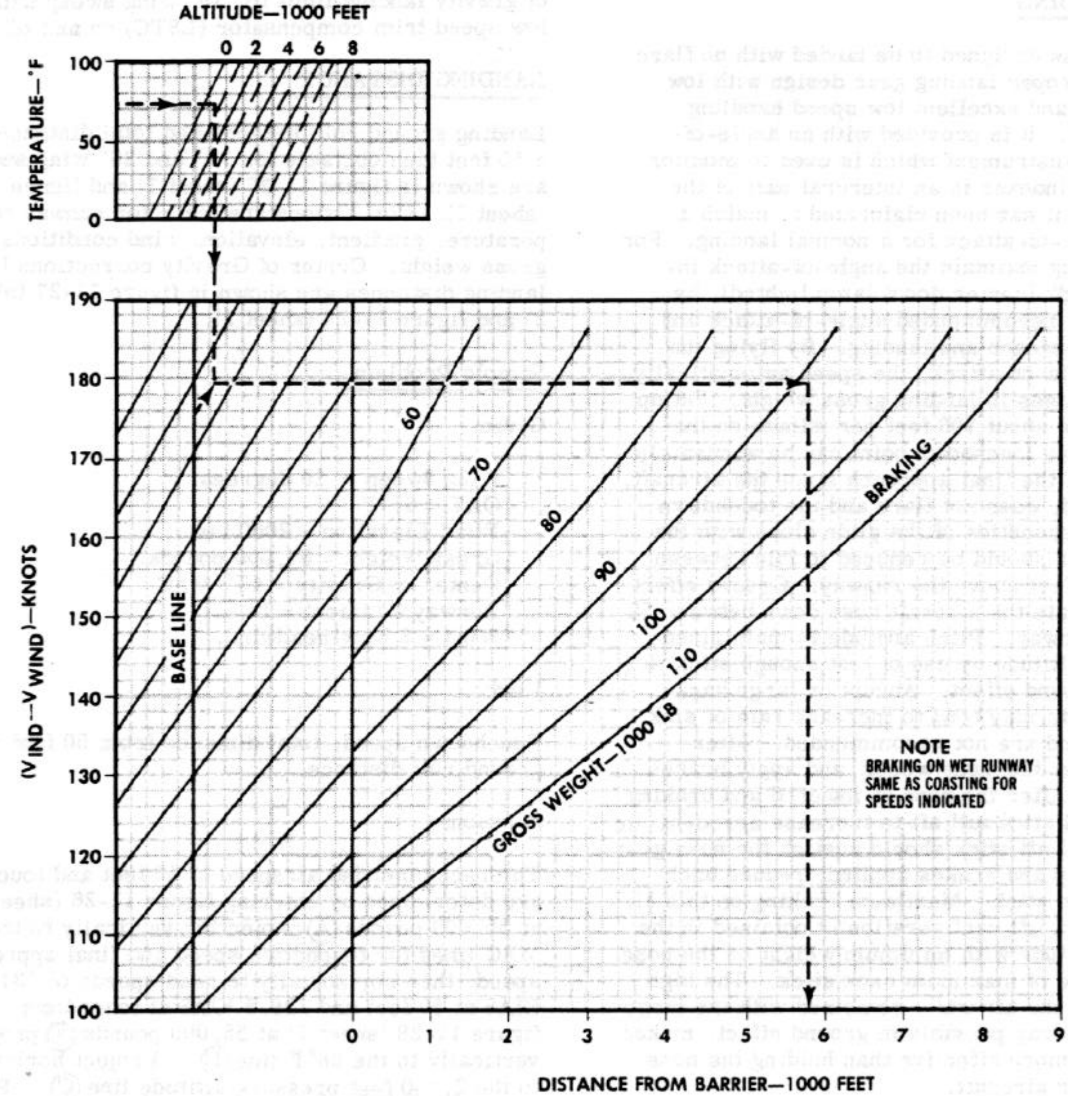

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Figure 11-25B (sheet 2)

part 5

LANDING

TABLE OF CONTENTS

NORMAL LANDING

The aircraft was designed to be landed with no flare by combining proper landing gear design with low landing speeds and excellent low speed handling characteristics. It is provided with an angle-of-attack indexer instrument which is used to monitor landings. The indexer is an intergral part of the aircraft in that it has been claibrated to match a speed and angle-of-attack for a normal landing. For a normal landing maintain the angle-of-attack indexer "on speed" (center donut lamp lighted), by maintaining the recommended angle-of-attack and airspeed for approach and landing. By flying the "on speed" angle-of-attack, the speed automatically adjusts for changes in landing gross weight. Set up a rate of sink of about 700 feet per minute on the glide slope. The touchdown point can be accurately estimated from the final approach since the aircraft, properly landed, does not flare and the touchdown point is the intersection of the glide slope with the runway. Power should be reduced to idle between 50 feet and 25 feet above the runway. Ground effect will tend to rotate the aircraft nose down between 50 feet and the runway. Pitch attitude is maintained below 50 feet altitude by use of just enough stick to counter the ground effect. Abrupt or large back-stick movements may tend to increase rate of sink momentarily and are not recommended. After touchdown lower the nose wheel, and apply brakes immediately. After the nose is lowered and braking is applied, pull stick full aft to increase aerodynamic braking. If full aft stick should unstick the nose gear, with spoilers up and brakes applied, reduce back pressure on the stick. Maximum braking on this aircraft in the high lift configuration is obtained in the three point attitude with minimum weight on the nose gear and by use of maximum back stick. The high aspect ratio of the aircraft, combined with the large horizontal tail drag possible in ground effect, makes this technique more effective than holding the nose high as in other aircraft.

LANDING SPEED CHARTS

Approach and touchdown speeds for the angle-of-attack indexer "on speed" indication for 16° and 26° wing sweep with full flaps are shown in figure 11-26 (sheet 1 and 2). Caution notes call out the aft center of gravity landing limit for 16° wing sweep with the low speed trim compensator (LSTC) on and off.

LANDING DISTANCE

Landing ground roll distance and total distance from a 50 feet high obstacle for 16° and 26° wing sweep are shown in figure 11-27 (sheet 1) and figure 11-28 (sheet 1). Corrections are made for runway temperature, gradient, elevation, wind conditions and gross weight. Center of Gravity corrections to landing distances are shown in figure 11-27 (sheet 2) and figure 11-28 (sheet 2).

Sample Problem

Given:

- Wing Sweep = 26 degrees.
- OAT = 68°F
- Field Elevation = 2000 feet.
- Gross Weight = 55,000 pounds.
- Center of Gravity = 45% MAC.
- Runway Gradient = 1%.
- Wind = 8 Knot headwind.

Find:

Touchdown speed, total distance from 50 feet and ground roll distance.

Solution:

The recommended airspeed at 50 feet and touchdown are determined by entering figure 11-26 (sheet 2) at 55,000 pounds (A) projecting vertically to the 45% MAC lines for touchdown speed and final approach speed, then horizontally to read speeds of 134.8 KIAS at 50 feet and 126.5 KIAS at touchdown. Enter figure 11-28 (sheet 1) at 55,000 pounds (A) project vertically to the 68°F line (B). Project horizontally to the 2,000 feet pressure altitude line (C). Project vertically to the wind effect base line (D), then follow parallel to the headwind line to 8 knots headwind (E). Project vertically to the runway gradient base line (F), then follow parallel to the downhill line to 1% gradient (G). Project vertically to read a ground roll of 1,600 feet for 40% MAC on hard, dry runway (H). Also read for a total landing distance from 50 feet above the runway, 2,350 feet (J).

To correct to 45% MAC, enter figure 11-28 (sheet 2) at the 40% MAC ground roll of 1,600 feet and the 40% MAC total distance of 2,350 feet, project vertically to the 45% MAC line, then horizontally to read a corrected ground roll of 1,500 feet and a corrected total distance from 50 feet of 2,225 feet.

EMERGENCY LANDING

In the event of a malfunction of some part of the high lift system (slats or flaps) or in the wing sweep drive system, it may become necessary to land the aircraft in any one of several configurations. In order to give landing information on as many situations as feasible, figure 11-29 (sheets 1 and 2) are presented. These data are shown for a range of weights which extend beyond the anticipated landing weights. It is highly recommended, however, that in emergency landings the aircraft weight should be reduced to the minimum acceptable by either consuming or dumping fuel. Figure 11-29 (sheet 1) has slats and flaps retracted in all cases and shows emergency speeds and distances for five different sweep angles. Figure 11-29 (sheet 2) has slats extended and deflected 45° in all cases, shows 26° sweep with various flap positions and 16° sweep with essentially full flap capability. The upper plot in each chart shows the recommended touchdown speeds, approach speed, and the pattern speed for maximum endurance. In some cases, at the high sweep angles, the governing factor on speed has been tail clearance which may dictate speeds at touchdown somewhat high than 115% stall speed landing. It should be noted that the currently recommended maximum braking speed line is often considerably below the touchdown speed. The plot immediately below this presents the total distances required if the aircraft rolls to maximum braking speed then applies brakes until the aircraft stops. Speeds at which the aircraft would engage an arresting cable with the tail hook under various conditions are shown in the plot in the lower right of each chart. The brakes are applied 5,000 feet from the cable in all cases in this plot, regardless of the aircraft speed at application. The distances here are the total distances from the point of touchdown to the point of arresting.

Sample Problem

Given:

Aircraft weight = 52,000 pounds.
Wing sweep = 72.5°.
Slats - Retracted.
Flaps - No extension or deflection.
Spoilers - Not activated for landing.
Runway length - 10,000 feet.

Find:

Pattern Speed.
(Recommended as the speed for minimum fuel consumption)

Touchdown speed.

Approach speed. (Add 10 knots)

Ground roll distance observing maximum braking speed.

Solution:

280 KIAS. PATTERN SPEED

200 KIAS. TOUCHDOWN SPEED

210 KIAS. APPROACH SPEED (SEE NOTE)

11,500 feet. GROUND ROLL DISTANCE WITH MAXIMUM BRAKING

It is obvious here that in this configuration and observing the braking limits, the aircraft requires more runway than is available by some 1,500 feet. If an arresting cable is available, however, we see from the lower right plot that by applying brakes at 5,000 feet from the cable the aircraft will be decelerated to approximately 58 knots when contact with the cable is made. This is well below the cable limit speed of 170 knots for this weight and will allow a safe landing.

FINAL APPROACH AND TOUCHDOWN SPEEDS

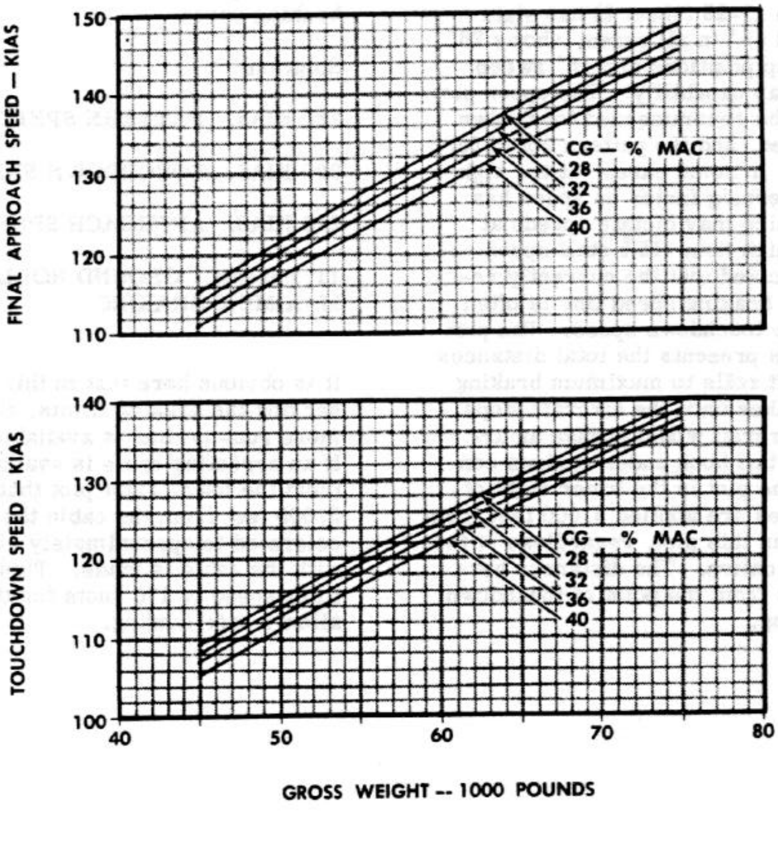
DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF 30-P-12
WING SWEEP = 16°
FLAPS: 37.5°
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

CAUTION

DO NOT LAND WITH 16° WING SWEEP IF THE CENTER OF GRAVITY IS AFT OF 39.5% MAC



26512-1/89-0

FINAL APPROACH AND TOUCHDOWN SPEEDS

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF 30-P-12
WING SWEEP = 26°
FLAPS: 37.5°
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

CAUTION

DO NOT LAND WITH 26° WING SWEEP IF THE CENTER OF GRAVITY IS AFT OF 56.5%MAC

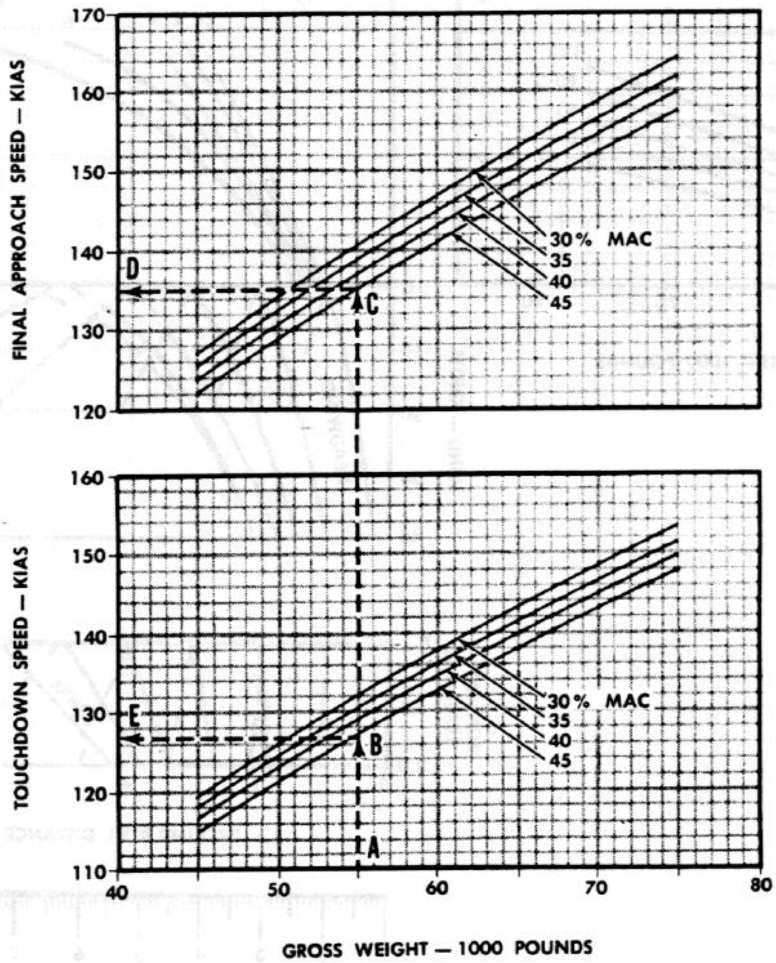


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Figure 11-26 (Sheet 2)

LANDING DISTANCE—16 DEGREE WING SWEEP

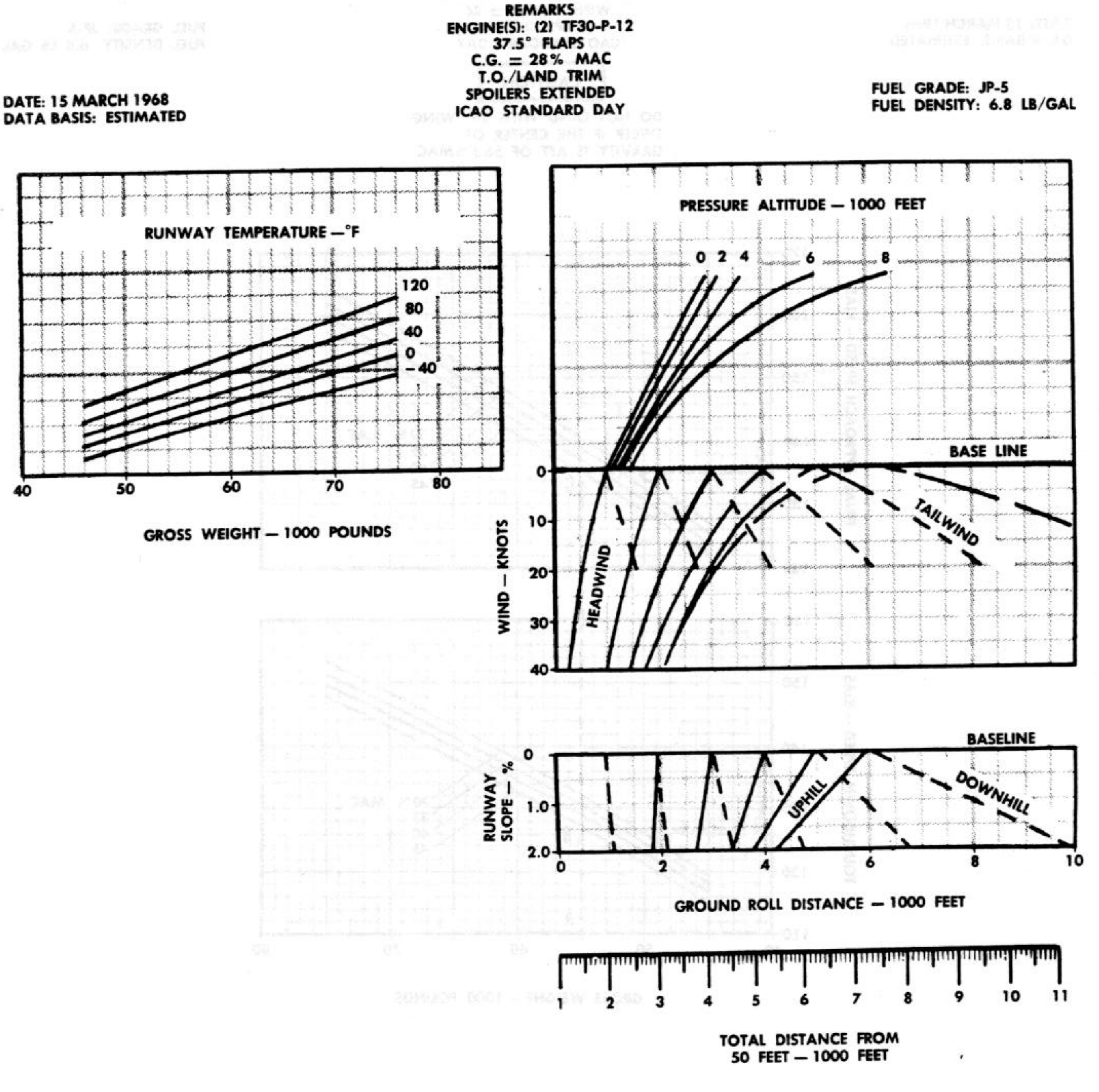


Figure 11-27 (Sheet 1)

LANDING DISTANCE—C. G. CORRECTION FOR GROUND ROLL OR TOTAL DISTANCE

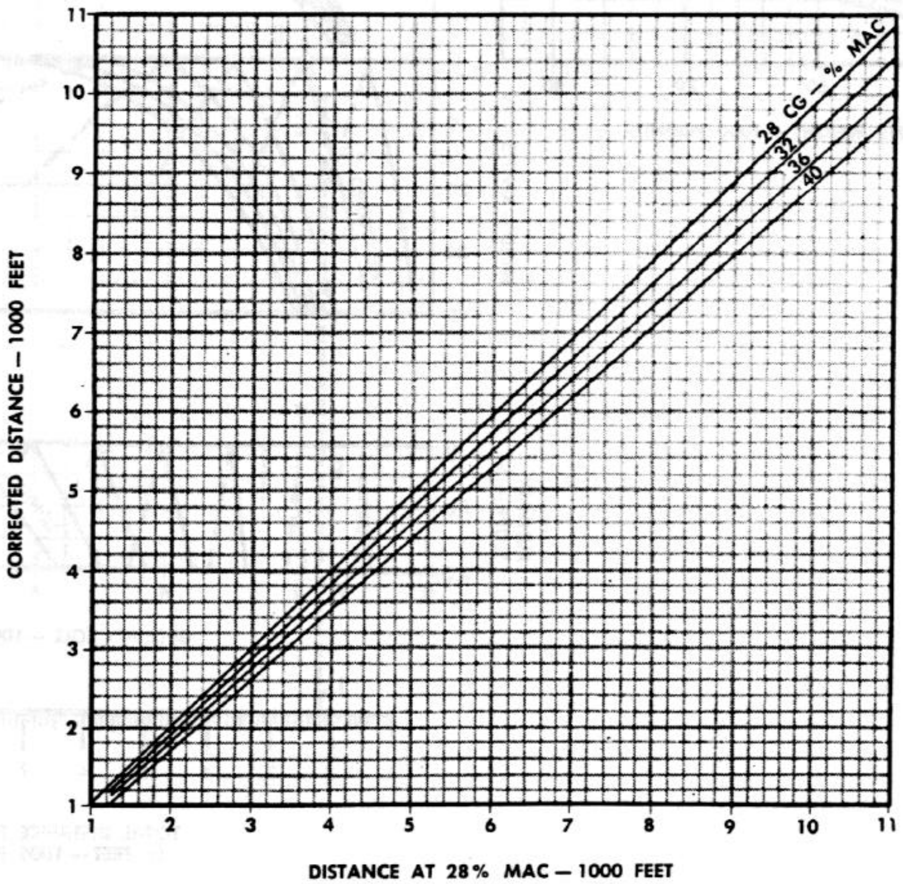
DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-12
WING SWEEP — 16°
FLAPS: 37.5°
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

CAUTION

DO NOT LAND WITH 16° WING SWEEP IF THE CENTER OF GRAVITY IS AFT OF 39.5% MAC



26512-1/91-0

Figure 11-27 (Sheet 2)

LANDING DISTANCE—26 DEGREE WING SWEEP

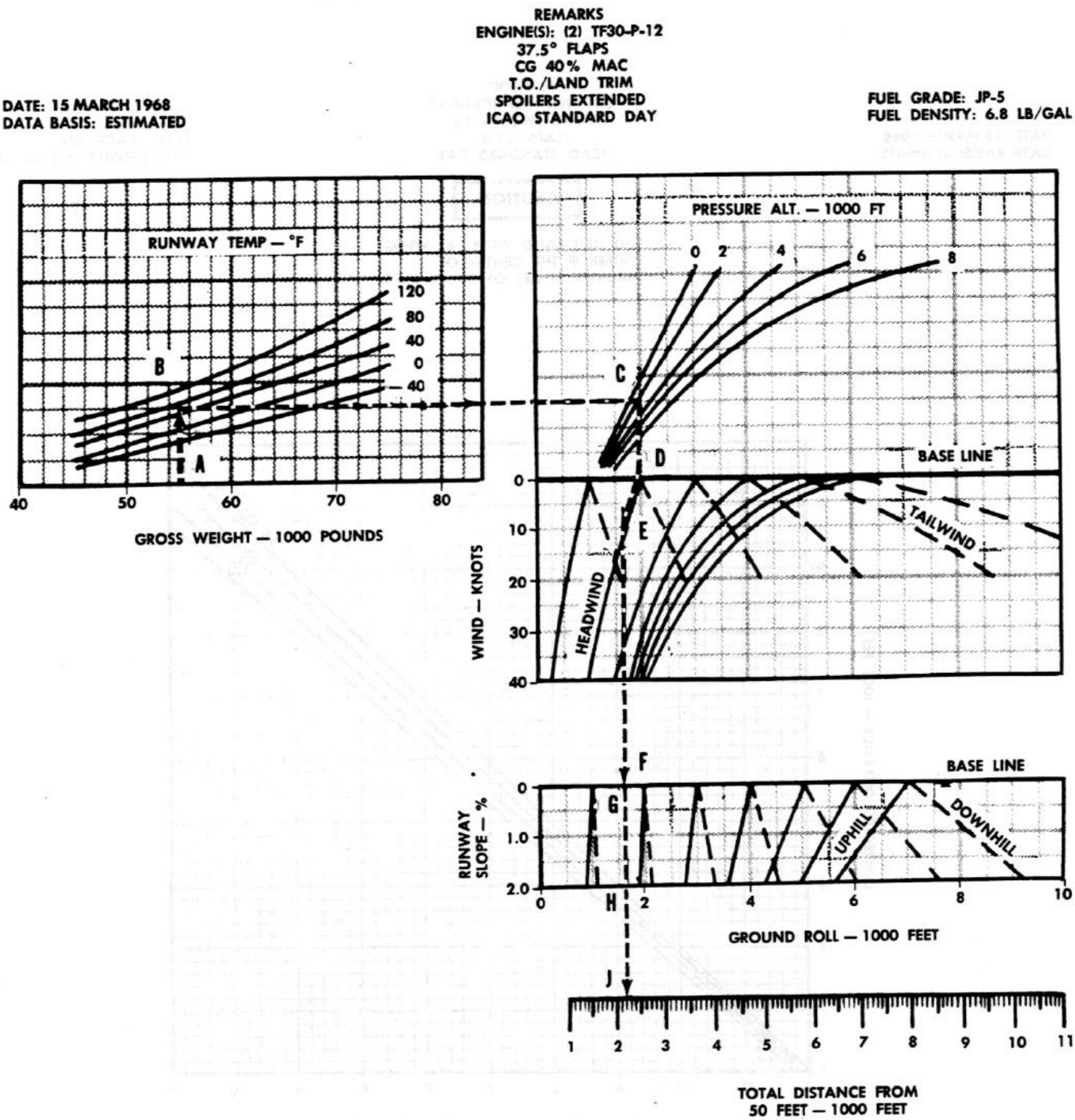


Figure 11-28 (Sheet 1)

LANDING DISTANCE—C. G. CORRECTION FOR GROUND ROLL OR TOTAL DISTANCE

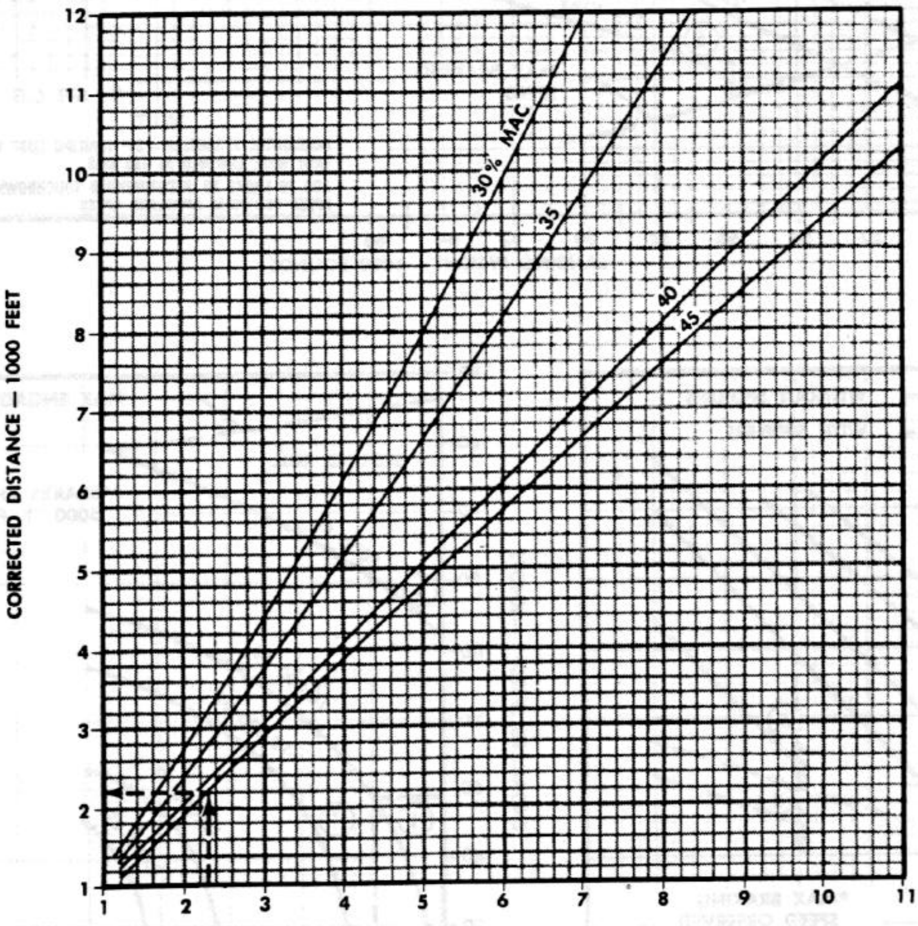
DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-12
WING SWEEP = 26°
FLAPS: 37.5°
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

CAUTION

DO NOT LAND WITH 26° WING SWEEP IF THE CENTER OF GRAVITY IS AFT OF 56.5% MAC



26512-1/94-0

Figure 11-28 (Sheet 2)

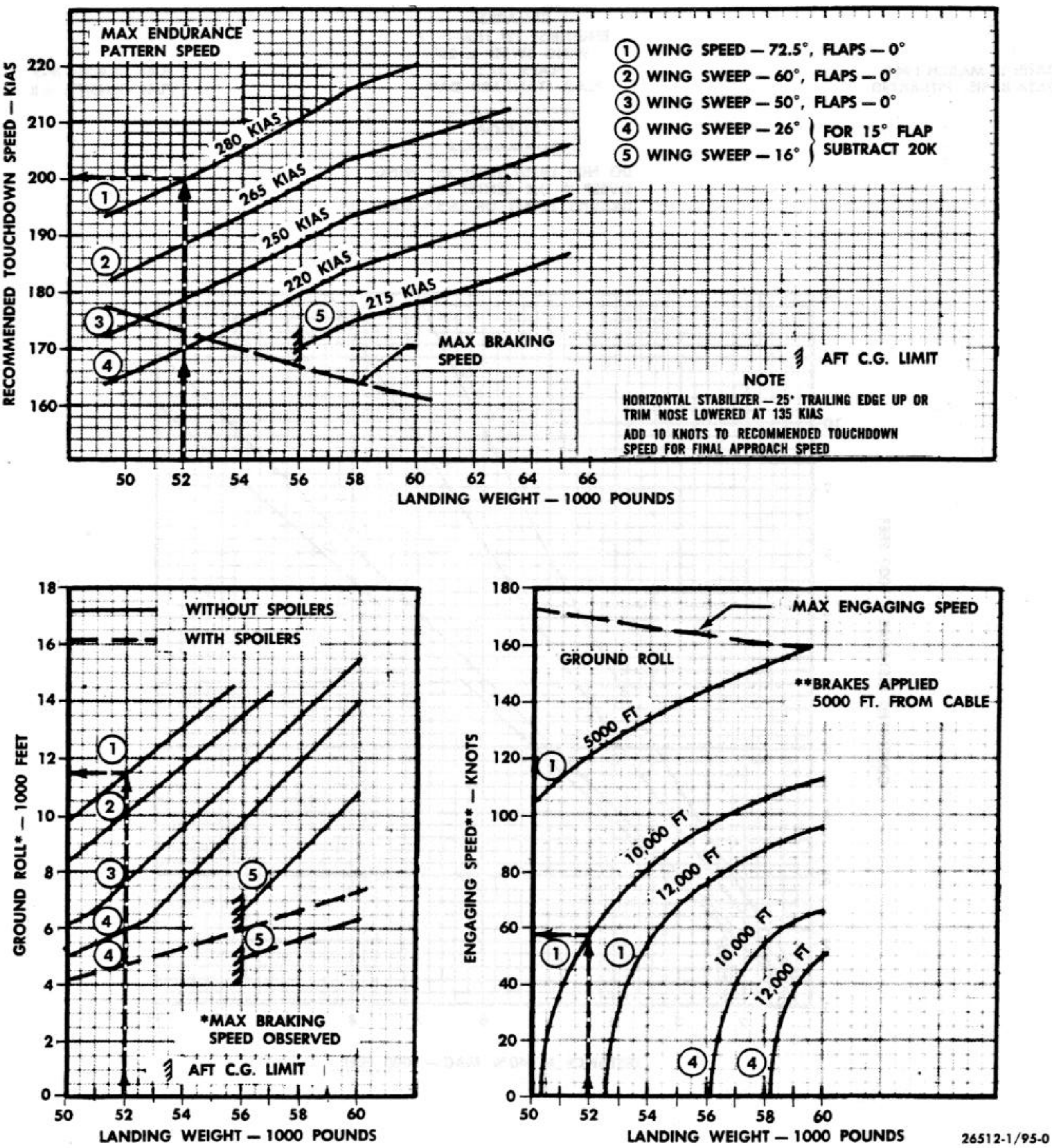
LANDING—EMERGENCY

ALTITUDE = SEA LEVEL

DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-12
CG = NORMAL SEQUENCE
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL



26512-1/95-0

Figure 11-29 (Sheet 1)

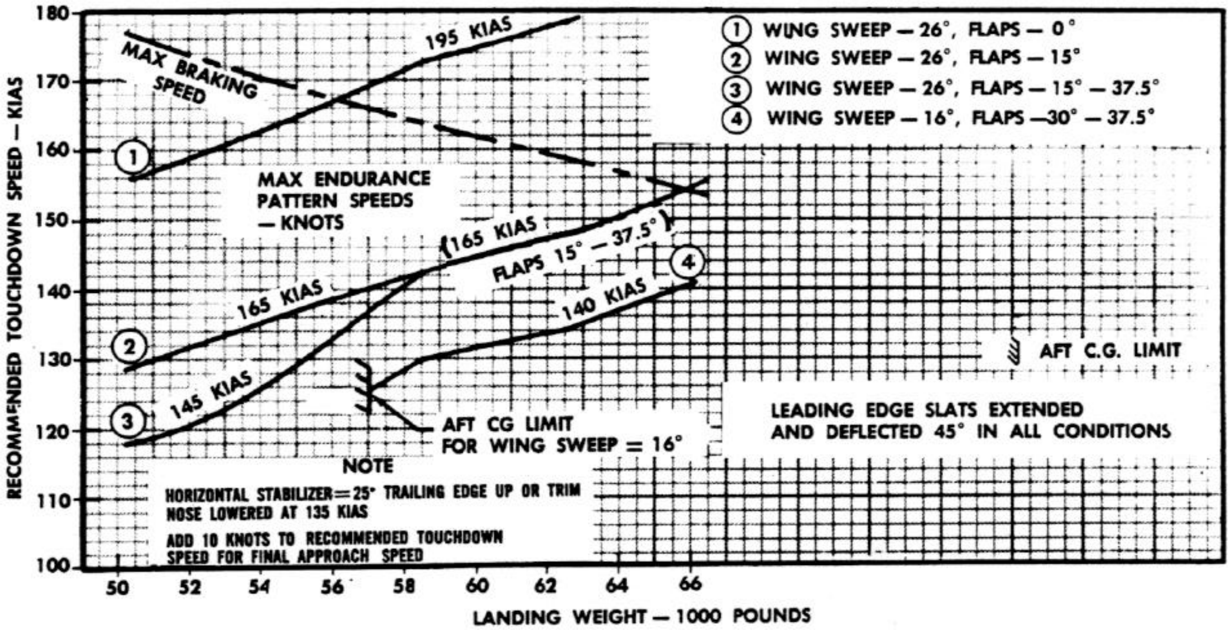
LANDING—EMERGENCY

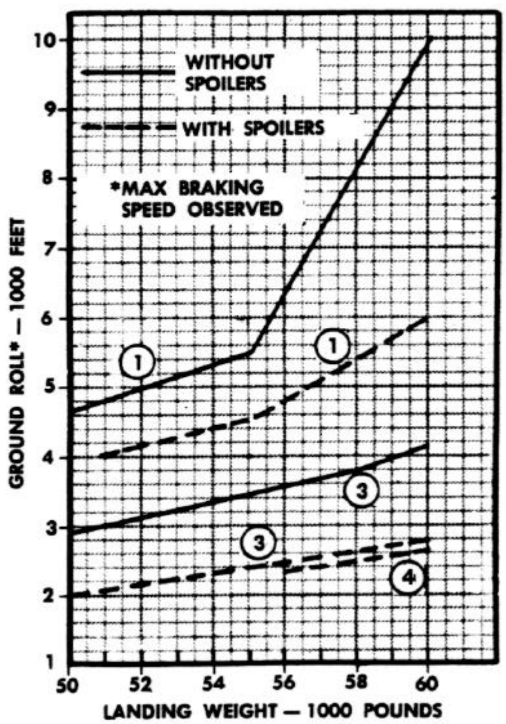
ALTITUDE = SEA LEVEL

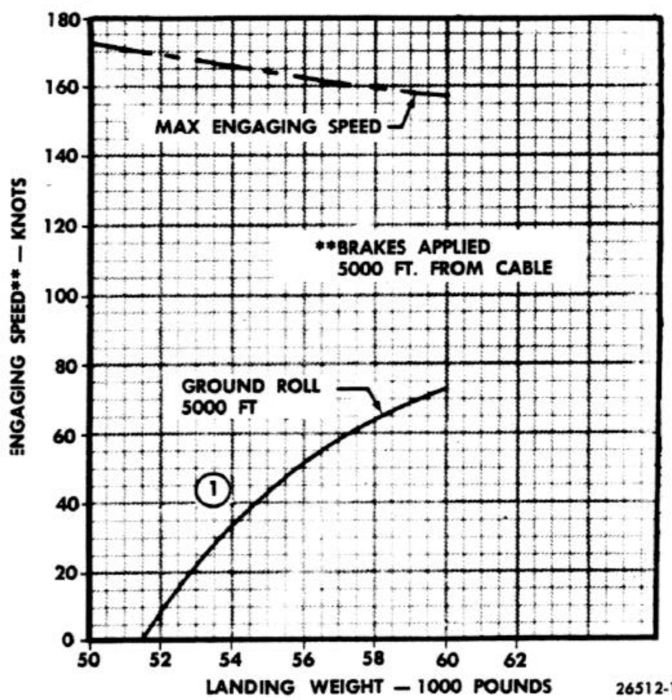
DATE: 15 MARCH 1968
DATA BASIS: ESTIMATED

REMARKS
ENGINE(S): (2) TF30-P-12
CG = NORMAL SEQUENCE
ICAO STANDARD DAY

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL







26512-1/96-0

Figure 11-29 (Sheet 2)

ALPHABETICAL INDEX

A

B

C

*DENOTES ILLUSTRATION

D

E

F

*DENOTES ILLUSTRATION

G

H

I

*DENOTES ILLUSTRATION

*DENOTES ILLUSTRATION

P

R

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*DENOTES ILLUSTRATION

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U

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W

Y

*DENOTES ILLUSTRATION